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SPECIAL HANDLING

FEASIBILITY STUDY

OF AN

AIR-LAUNCHED, SINGLE-PASS, LOW-ORBIT

RECONNAISSANCE SYSTEM

SPECIAL HANDLING

TABLE OF CONTENTS

Approved For Release 2001/08/28 : CIA-RDP66B00762R000100070001-6

SUMMARY

This study describes a reconnaissance satellite system which will be air launched at an altitude of 80,000 feet at a speed of Mach 3 and solid propellant boosted into single pass, low altitude, nearly circular orbits which will cover any target in the denied area and permit aerial recovery of the re-entry capsule, at Johnston Island, with minimum impact dispersions. This reconnaissance system will be capable of photographing 150,000 square miles with a 5' ground resolution at an altitude of 80 nautical miles completely independent of any ground control.

Three vehicle configurations are discussed. Two of these (Configurations B/B' and C) utilize the Polaris A-3 motors currently under development. Three payload systems, logical extensions of the Corona Program, have been considered. Two configurations (I and II) are direct modifications of this subsystem; the third (Configuration III) is a direct application of a proposed single-lens stereo camera. Prime effort has been directed toward payload Configuration I and vehicle Configuration B/B'. Payload-vehicle compatibility at the 80 mile orbit altitude are listed below. Although Configuration B (current estimated A-3 motor performance) is capable of carrying only payload Configuration II, current motor improvement efforts could lead to Configuration B' with a capability exceeding that required for payload Configuration I.

Vehicle Configuration (Capability) \ Payload Configuration (Weight)	I (456 lbs.)	II (300 lbs.)	III (700 lbs.)
A (635 lbs.)	X	X	
B (340 lbs.)		X	
B' (610 lbs.)	X	X	
C (1010 lbs.)	X	X	X

The inertial guidance and control subsystem steers the vehicle precisely into the pre-programmed orbit with accurate attitude references to limit "over target" cross track position error to a maximum of 3 miles. Proper attitude and timing references are provided for the orbital and capsule re-entry phases; with this subsystem, re-entry impact dispersions are only 20 miles uprange and 25 miles downrange with crossrange dispersions in the order of 5 to 7 miles.

The recovery subsystem is a direct adaptation of the light-weight re-entry capsule developed for the ML-470 Program which permits aerial recovery (prime) or water/land pickup (secondary).

It is concluded from this study, therefore, that an air launched reconnaissance satellite system, capable of keeping pace with the Corona Program, can be designed around hardware components being developed on existing programs and will meet the established mission objectives.

INTRODUCTION/OBJECTIVES

This study has been undertaken to evaluate the feasibility of a low-altitude orbital reconnaissance system to be launched from an aircraft capable of operating from normal military bases. The space vehicle is launched from the airplane at M = 3 and 80,000 feet at not more than 1000 miles from the take-off site.

The mission objectives are to photograph, from a low-altitude orbit, any target within the denied area and recover the data at Johnston Island. Ground resolution is to be 5 feet and the mission life is one orbit.

FLIGHT PROFILE AND SYSTEM DESCRIPTION

The system and flight sequence from take-off to recovery is described. The specific mission discussed is a single-orbit operation with recovery at Johnston Island, southwest of the Hawaiian Islands. Fundamentally the concepts are the same for similar operations with recovery sites inside the continental U.S., with the possible exception of more critical range-safety constraints and necessarily a different range of optimum orbit inclinations.

Take-Off and Launch

The carrier vehicle would take-off from a field in the Hawaiian Islands, for example Hickam A.F.B. on Oahu, and cruise at an unspecified speed and altitude to the launch site, 250 to 600 miles to the east. This will be covered later in the discussion of Figures 2 and 3. This study has assumed that the cruise will be at low enough speed that no critical high temperature environment is imposed upon the space system;(if the temperatures are high, heat protection provisions would be necessary, but this will not impair the feasibility conclusions of this report). At the launch site the carrier will attain the prescribed conditions for launch, namely M = 3.0 at 80,000 feet at the optimum flight path angle for launch. Carrier angle of attack has been assumed to be small.

The space system is provided with an aft extension with aerodynamic fins to provide static stability during the free-fall period. The vehicle is completely passive during this phase. Interference effects have not been considered here and free fall times have been assumed short enough to have a negligible effect on space vehicle performance.

Ascent

At or just prior to first stage ignition the aft fairing will separate from the vehicle, and the vehicle will be controlled by the engine thrust vectoring system after ignition.

The vehicle will fly a pre-programmed trajectory to the specified orbit under the control of the inertial guidance system in the orbital vehicle stage. First stage profile is a gravity turn trajectory whose coast apogee is approximately the altitude of the desired orbit. At burnout, the first stage is separated; the vehicle is oriented to the horizontal by the orbital stage cold gas system during the coast phase. Second stage ignition occurs near the coast apogee and the vehicle attains orbital velocity.

When orbital velocity is attained, the guidance system commands thrust termination and separation of the spent motor.

Orbit

After separation from the second stage motor the orbital vehicle (OV) is turned 90° to the orbital plane to accommodate the payload. The guidance system determines the final orbit parameters after all forces including second stage engine tail-off have decayed, computes the appropriate times to turn on the payload, reorients the OV and initiates capsule separation for recovery at Johnston Island. During payload operation the guidance system programs the small yaw corrections to compensate for the earth's rotation. Vehicle attitude control is provided by a reaction control system utilizing stored nitrogen as the working fluid.

As the OV approaches the point for de-orbiting, it is oriented to place the re-entry capsule at the retro angle for optimum recovery (minimum re-entry range and minimum impact dispersion) and the capsule is separated from the OV.

Capsule De-orbit and Re-entry

The capsule is spun up to approximately 100 RPM to provide body stability during burning of the retro motor. The retro-burn continues to propellant exhaustion, when the capsule has been accelerated to approximately 1300 ft/sec (relative to the OV). Atmospheric re-entry and deceleration of the stabilized capsule is terminated by parachute release as subsonic speeds are reached. The suspended capsule is then recovered by aircraft during parachute descent.

SYSTEM CONSTRAINTS AND ASSUMPTIONS

Certain carrier-airplane conditions have been imposed and other assumptions have been made. These are listed below.

Launch Vehicle - Given:

Capable of launching at M = 3.0 at 80,000 feet. Launch angles to 30° with a possibility of going to as much as 45°. Low angles of attack. 0°-15° The airplane radius is approximately 1000 miles. Guidance errors after two hours of flight are 100 seconds in azimuth and 1-2 miles in position.

Assumed:

No interference flow field exists, that is it has been ignored. Appropriate structural tie points to the space vehicle will be provided. The aerodynamics of the space vehicle will not prohibit attainment of the launch conditions mentioned above. The aircraft will cruise to the launch point at speeds low enough not to expose the space vehicle to high temperatures.

Space Vehicle - Given:

Vehicle length shall not exceed 30 feet and the depth below the airplane shall not exceed 40 inches (these requirements have been relaxed to 35' and 54 inches). No gross weight limitation.

Assumed:

An aft extension or fairing to contain dropable aerodynamic stabilising surfaces is not part of the 30-foot length restrictions if it is contoured so that it will not impose any further restrictions on the aircraft at take-off (angle at "unstick").

The Mission - Given:

Single orbit with recovery at Johnston Island (originally the southwest U.S.). Orbit inclination flexibility to permit passage over any part of the denied area. High recovery impact accuracy.

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OPERATIONS STUDY

The relationship of launching site to recovery location for limited-pass missions has been studied extensively during the last ten months. The significance of the so called "conjugate point" for Johnston Island and the effect upon launch operations is discussed. Only prograde orbits are considered since the degradation of weight capability for westerly launches is unacceptable and the prograde orbit results in the vehicle desirably entering the denied areas from the south and west.

The "Conjugate Point"

If the traces of a family of orbits with different inclinations but the same period are made to intersect at a given location on the surface of the earth they will intersect again on the next pass at the same latitude but displaced in longitude by the angular distance the earth has rotated under the orbit during that pass. These points of intersection are called conjugate points. The concept is not valid for more than a very few passes since the error due to the rotation of the orbital plane (because of the earth's oblateness) can become significant. Complete flexibility of azimuth selection can be provided if the satellite is injected into orbit at a conjugate point of the recovery site.

For recovery at Johnston Island of orbits whose period is 87.3 minutes, the launch site should be 21.8° to the east or at 147° 40' W. with the same latitude of 17° N. The maximum shift for a 30° orbit due to the rotation of the orbit plane is approximately 28 miles for the one pass; the effect is less at higher inclination angles.

Other Recovery Sites

Similar reasoning can be applied to other recovery locations. Two cases are discussed briefly.

1. Southwestern U.S. Recovery. Launch conjugate is in the south-eastern United States and the inclinations tend to be more polar to the north and south to about 50° prograde in each direction. The orbiting vehicle would enter the denied area from the North on passes over Western Russia and East Europe. Recovery of visual payloads would occur at night. Range safety considerations would require extensive aircraft facilities near the north Atlantic seaboard, and possibly in Canada and South America.

2. "Baby Flattop". It has been suggested that a small aircraft carrier be used as a site for recovery operations.

The number of launch sites which will permit complete inclination capability should increase. Recovery in daylight is enhanced west of Johnston Island. Recovery can occur in the southern hemisphere, but the limitations of orbit passage into the denied area will be the same as an Atlantic Ocean or U. S. recovery.

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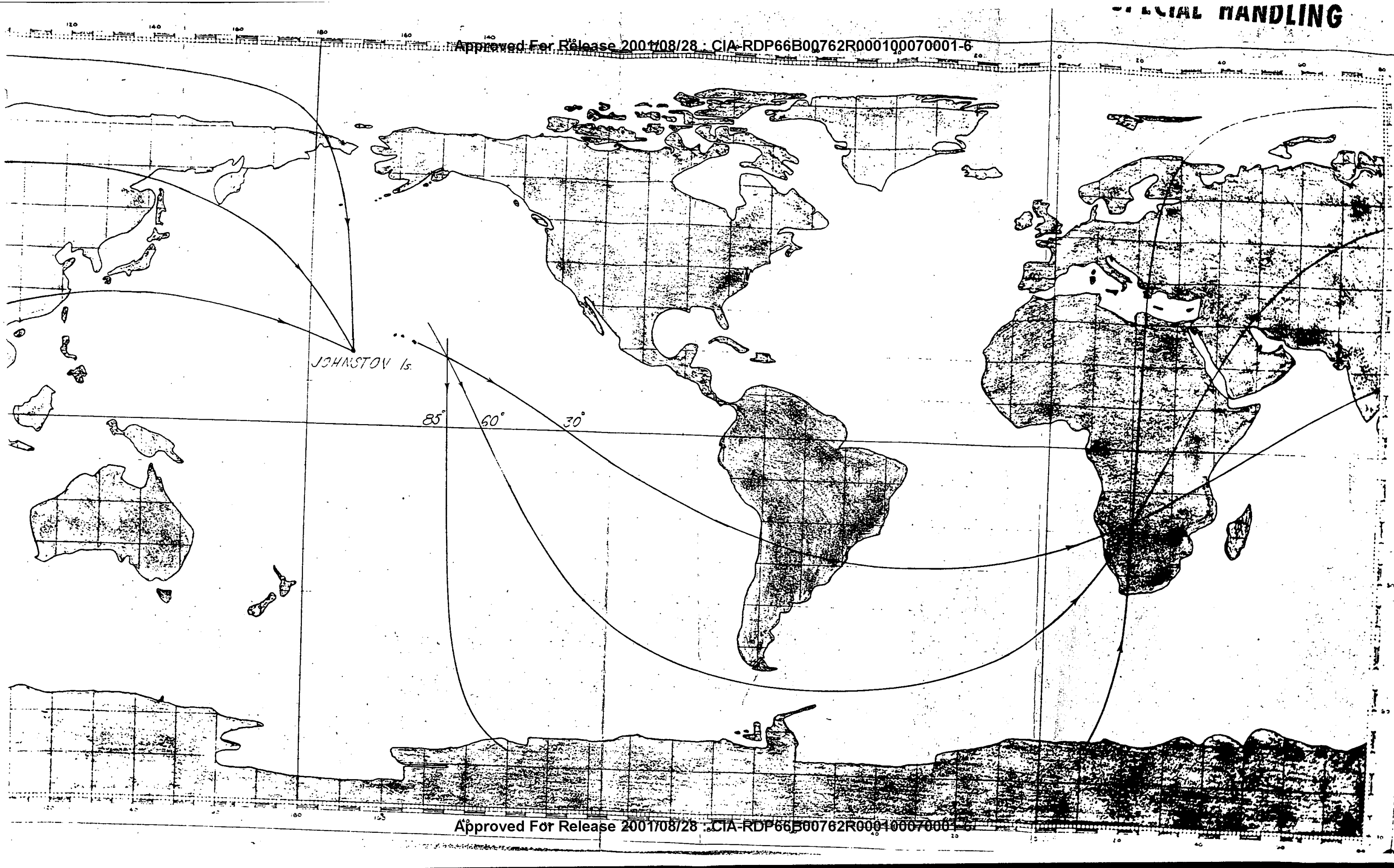

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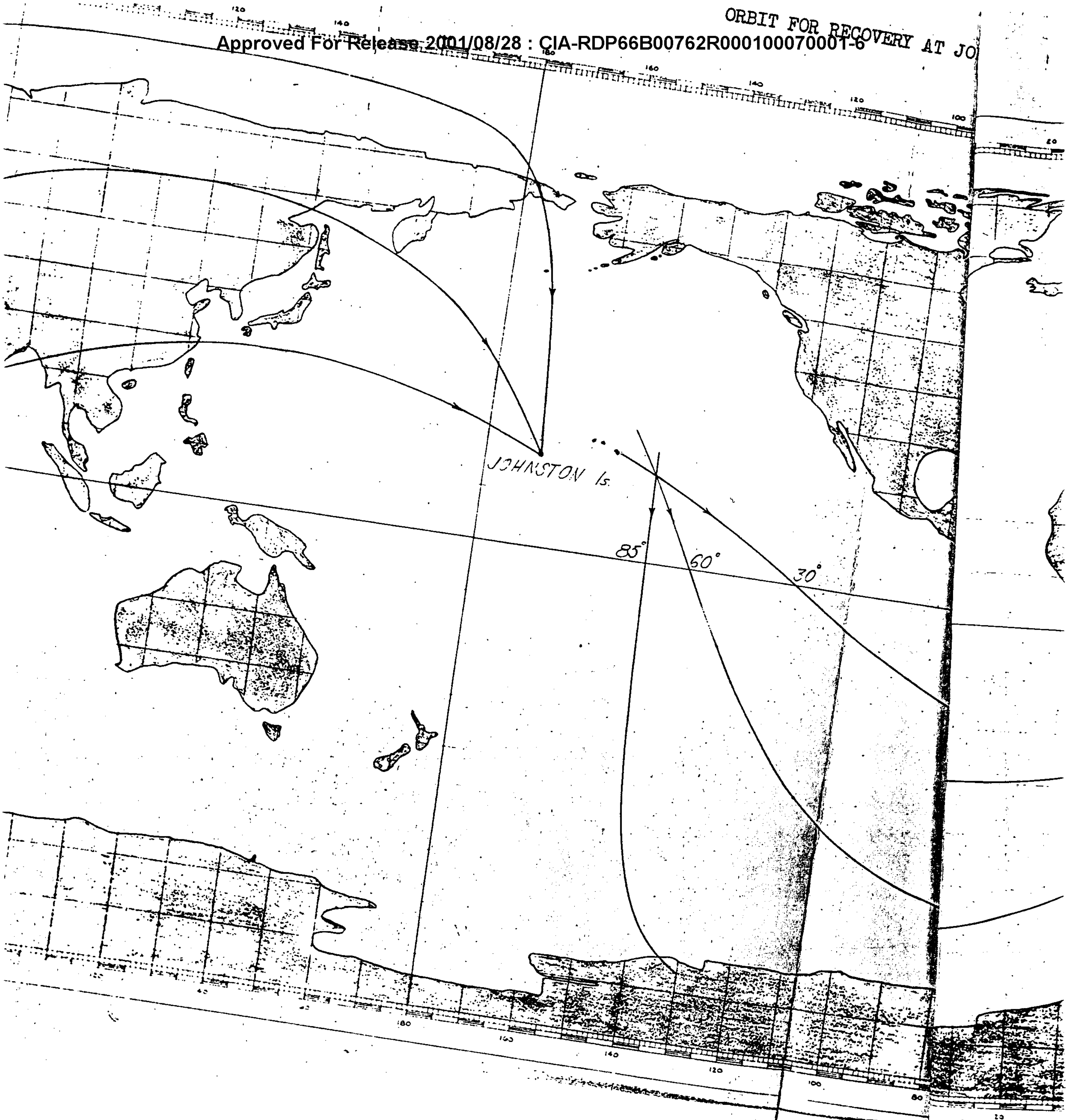

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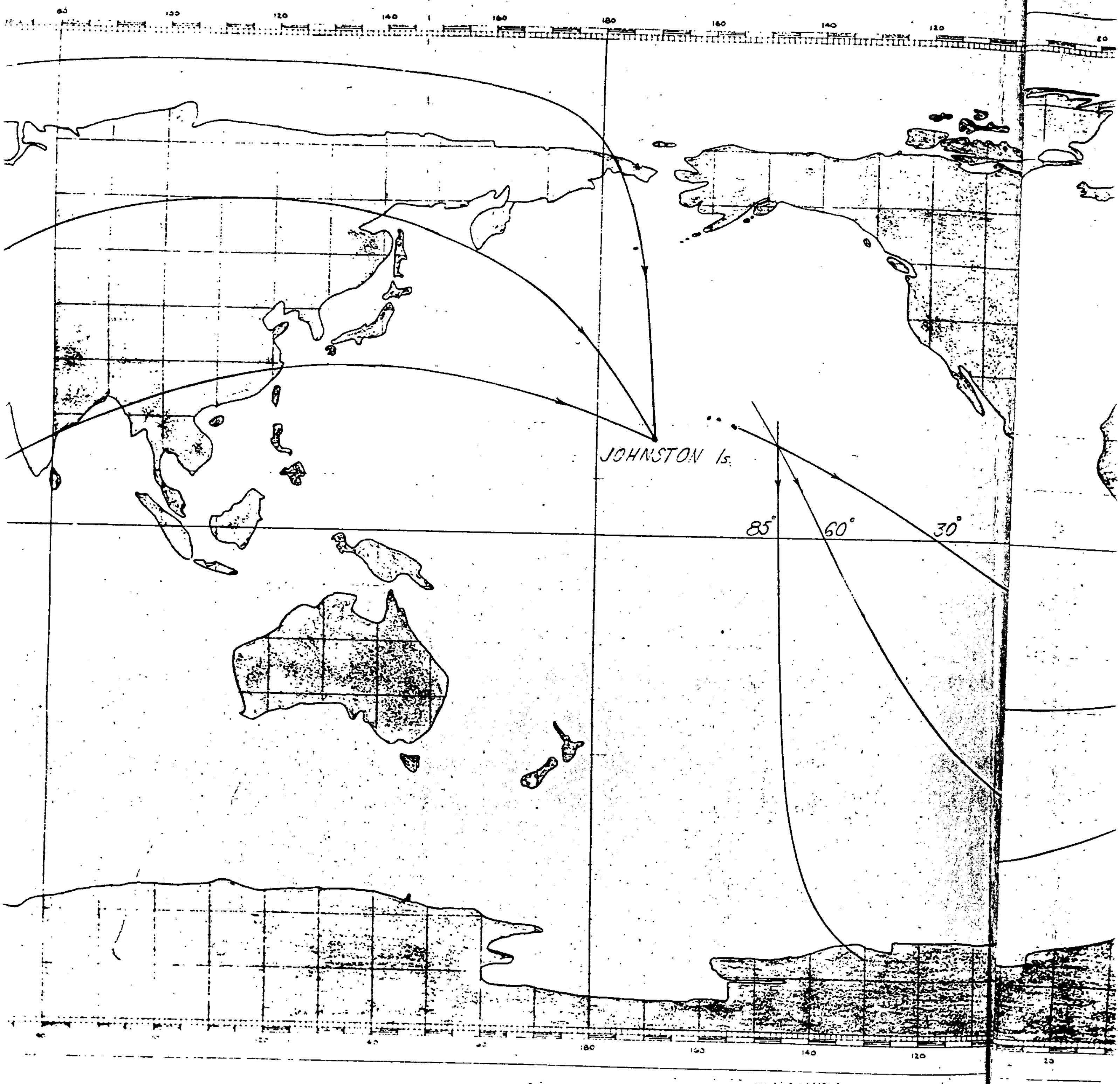

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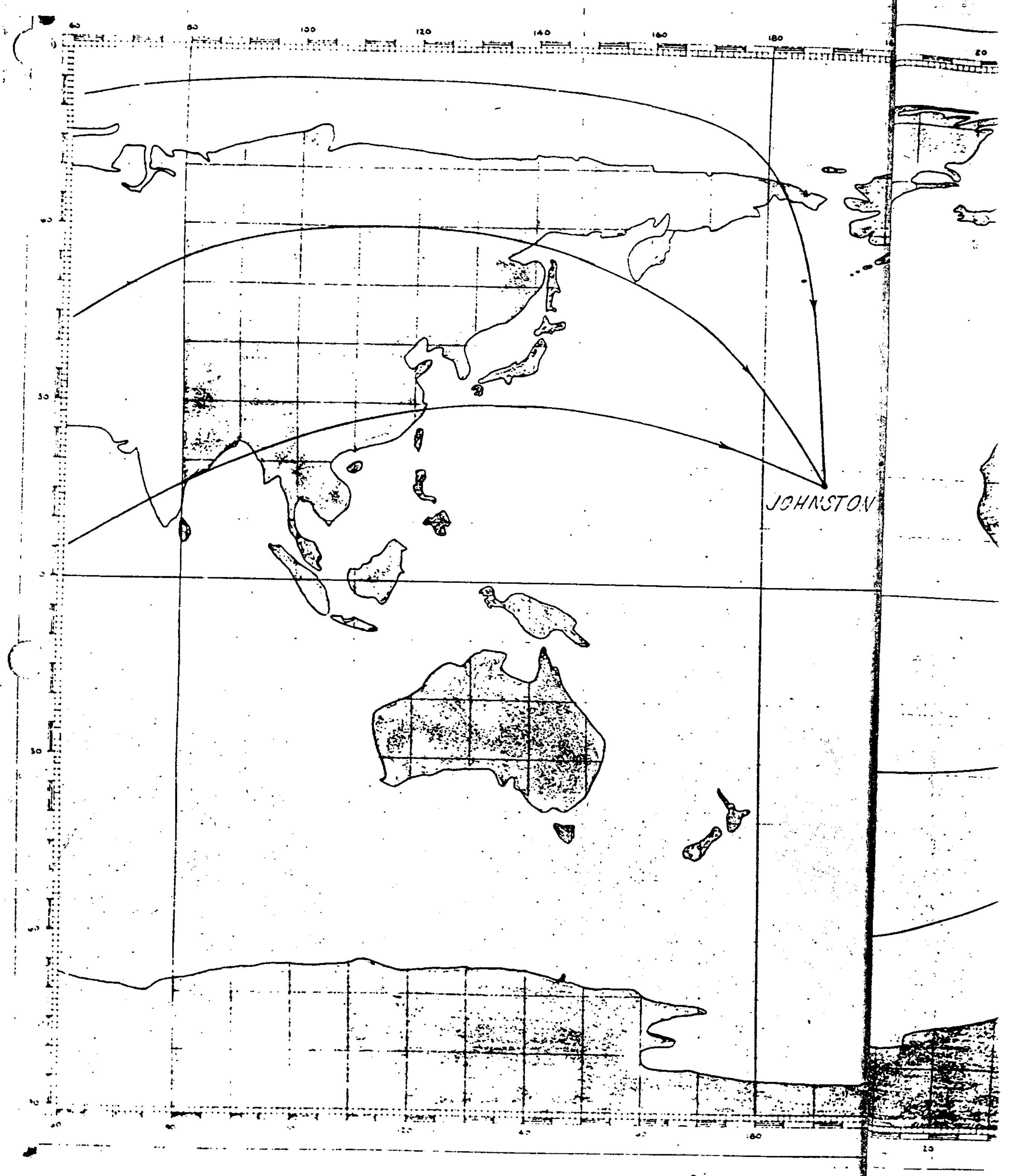

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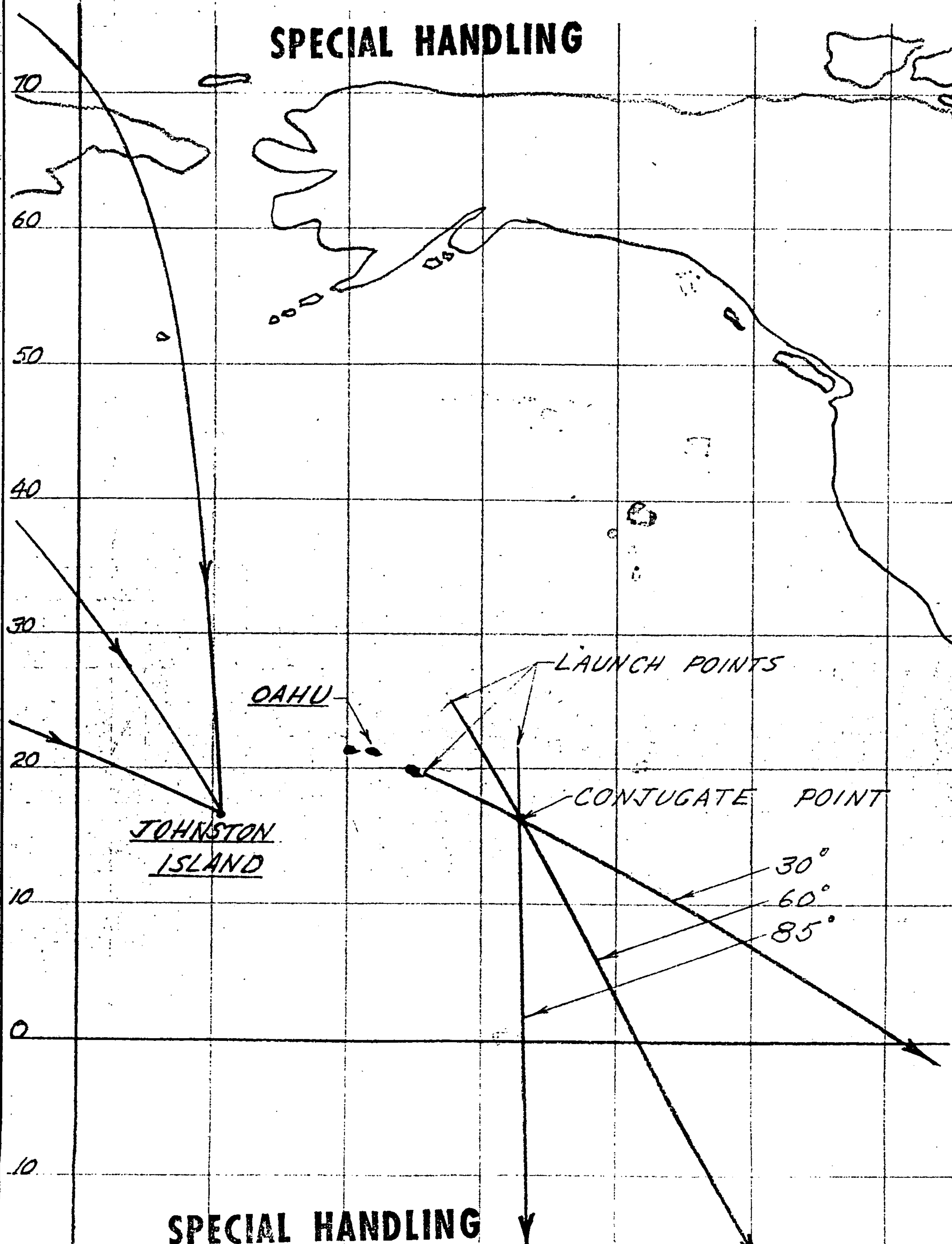

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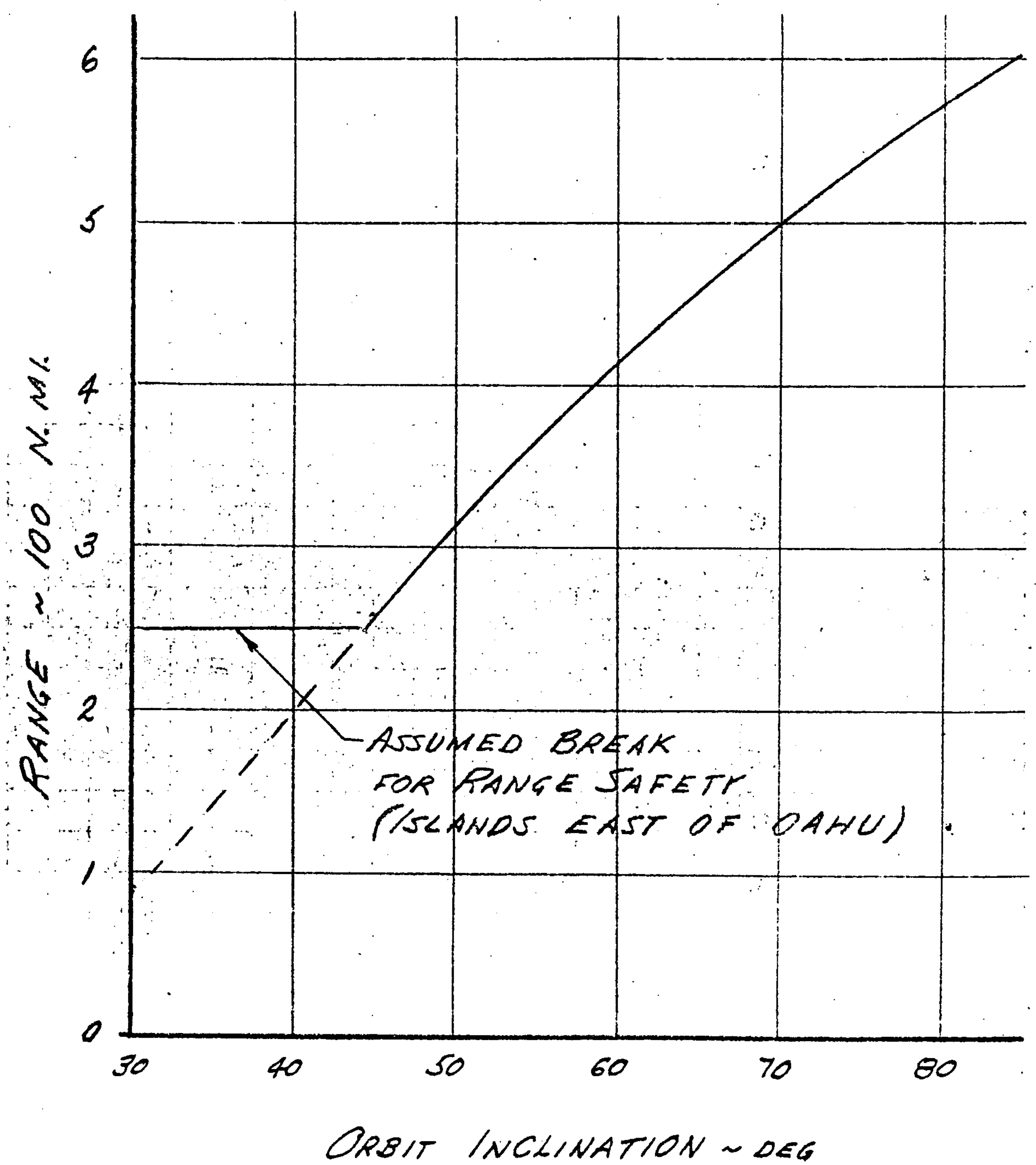
MINIMUM AIRCRAFT RANGE

HICKAM AFB TO LAUNCH SITE



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FORM MSD 362B

FIGURE 2

VEHICLE SYSTEM

CONFIGURATION A

Configuration A presents a method for achieving the desired performance capability within the restrictions of a 40" vehicle depth limitation and a 30 foot length limitation. The dual first and second stage motors, as proposed by LPC, result in a vehicle width of 80". The resultant payload and equipment-bay module is 40" x 80" in cross section and approximately 71" long.

Relaxation of the 40" depth limitation led to a more detailed study of the application of the 54" diameter Polaris A3 propulsion system as described in 'Configuration B'.

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CONFIGURATION A
40 x 80 System

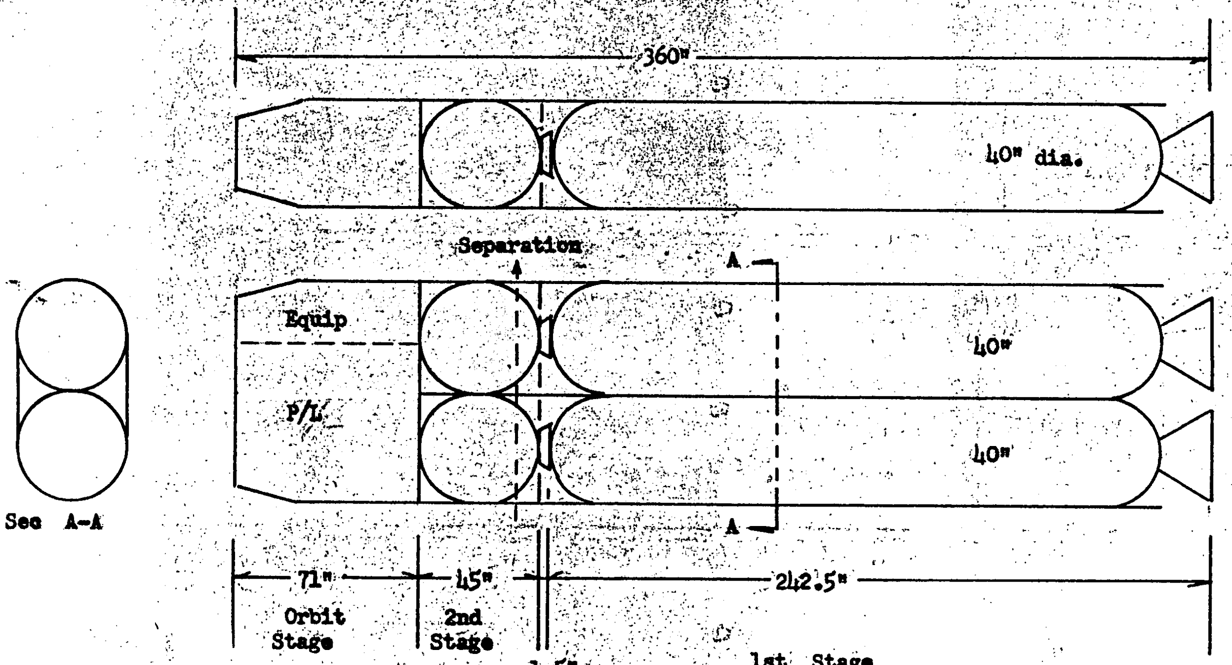

FIGURE 4

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WEIGHT STATEMENT
Configuration A

FIRST STAGE			29,582
Propulsion		28,432	
Propellant	26,000		
Inert (200 Expendable)	2,432		
Ballistic Shell (Interstage)		100	
Flight Controls		950	
TVC Hardware	200		
TVC Fluid	750		
Electrical (Wiring & Conduit)		50	
Contingency		50	
SECOND STAGE			5,320
Propulsion		5,000	
Propellant	4,500		
Inert (90" Expandable)	500		
Ballistic Shell (Interstage)		100	
Electrical Harness & Conduit		25	
Flight Controls		170	
TVC Hardware	80		
TVC Fluid	90		
Contingency		25	
ORBITAL VEHICLE (Less P/L Module)			340
Structure (Equip. Bay) 40 x 20 x 70		100	
Electrical		70	
Guidance		100	
Attitude Control		50	
Destruct		10	
Contingency		10	
NOSE FAIRING			84
GROSS WEIGHT (Less Payload Module)			35,326

WEIGHT SEQUENCE
Configuration A

GROSS WEIGHT (Less P/L)	35,326
Ignition Losses	12
FIRST IGNITION (Less P/L)	35,314
Propellant	26,000
Inert Expended	200
TVC Fluid Expended	700
FIRST B/O (Less P/L)	8,414
First Stage Separation	2,670
Nose Fairing	84
Second Stage Ignition Losses	10
SECOND IGNITION (Less P/L)	5,650
Propellant	4,500
TVC Fluid	80
Inert Expended	90
SECOND B/O (Less P/L)	980
Less Second Stage Inert	640
WEIGHT ON ORBIT (Less P/L Module)	340

CONFIGURATION B

Configuration B is an adaptation of the Polaris A3 missile. The A3 first stage is unmodified except as required for attachment to the carrier, and is complete with the necessary flight controls and electrical power system. The interstage structure configuration between the A3 first and second stages is also maintained. The second stage motor is modified by the addition of thrust termination ports and forward skirt. The liquid injection thrust vector control system is maintained. The A3 second stage forward equipment bay and equipment is deleted and a new equipment bay structure is added. The new equipment bay, 54" in diameter and 18" long, is part of the orbiting vehicle, and incorporates a forward ring for attachment of the payload module and an aft ring separation joint for separation from the second stage motor. The equipment bay, which may be manufactured as a separate module, contains the guidance system, the on-orbit attitude control system, a payload destruct system, and the electrical power system.

The proposed inertial guidance system supplies commands to the first-stage hydraulically-actuated control system and to the second-stage thrust-vector control system during ascent, and contains a pre-selected program for on-orbit operation. During the orbital phase, the guidance system supplies the information for attitude control, roll and yaw steering, payload operational sequencing commands, and re-entry orientation and capsule separation commands for the initiation of the capsule re-entry phase.

The on-orbit attitude control system components are current Agena hardware. A 1728 in^3 pressure bottle supplies adequate control gas storage to meet mission requirements plus an approximate 50% margin.

The electrical power system also uses Agena hardware components. The suggested type VI battery provides a large safety margin for the guidance and control systems power requirements.

The payload module length allowance with this system is approximately 70" when the vehicle overall length is restricted to 30 feet.

An aerodynamic nose fairing and aft extension complete the vehicle configuration. The aft extension fairing incorporates horizontal and vertical aerodynamic surfaces to provide stability during drop from the carrier and the entire system is jettisoned prior to ignition of the first stage. The nose aerodynamic fairing configuration will be designed for compatibility with the carrier configuration.

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CONFIGURATION B

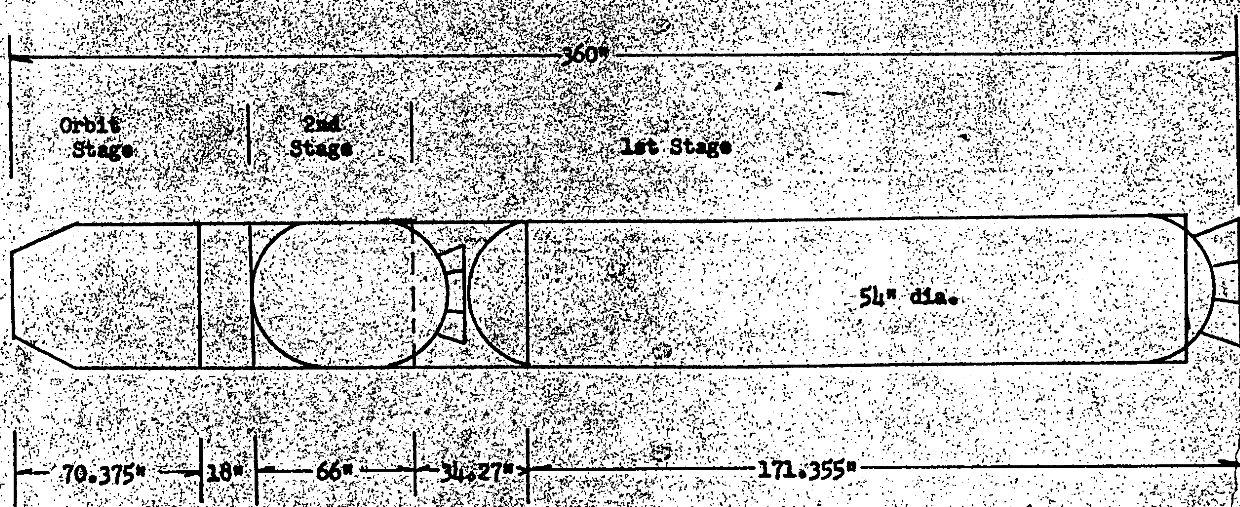

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FIGURE 5

CONFIGURATION B

EQUIPMENT LIST & WEIGHTS

FIRST STAGE				24,096
Propulsion			23,680	
Propellant		20,730		
Inert		2,950		
Chamber & Skirts	943			
Nozzles	1,421			
Insulation	468			
Ignition System	37			
Heat Shield	39			
Paint & Misc. Hardware	42			
Ballistic Shell			117	
Basic Structure		83		
Windows & Doors		10		
Equipment Supports		6		
Primacord Ring		4		
Insulation		12		
Paint		1		
Conduit Fairing		1		
Electrical			110	
Interlocks		6		
Harnesses		19		
Conduit		35		
Exploding Bridge Wire System		5		
Battery		31		
Power Distribution		12		
Umbilical Connector		2		
Flight Controls			154	
Motor Pumps		28		
Actuator Assemblies		64		
Structure		4		
Oil		3		
Transfer Valves		3		
Cover Assemblies		52		
Contingency			35	

CONFIGURATION B

EQUIPMENT LIST & WEIGHTS

SECOND STAGE				9,873
Propulsion			9,580	
Propellant		8,880		
Inert		700		
Chamber & Skirts	225			
Nozzles	170			
Thrust Termination Ports	80			
Insulation	170			
Ignition System	17			
Aerodynamic Heat Protection	7			
Base Heat Protection	18			
Paint & Misc. Hardware	13			
Ballistic Shell			11	
Forward Separation Ring		11		
Electrical			31	
Harnesses		8		
Conduit		23		
Flight Controls-TVC System			226	
Tanks & Supports		36		
Plumbing & Valves		32		
Pressurization System		24		
Manifold & Harness		14		
Fluid		120		
Contingency			25	

CONFIGURATION B

EQUIPMENT LIST & WEIGHTS

ORBITAL VEHICLE - Less P/L			290
Structure - Equipment Bay		50	
Skin & Doors	21		
Forward Ring	5		
Aft Ring	5		
Equipment Supports	15		
Insulation	4		
Electrical		70	
Battery - Type VI A	26		
Power Distribution	10		
Harnesses	34		
Inertial Guidance System		100	
Attitude Control System		50	
Control Nozzles & Valves	8		
Pneumatic Plumbing	4		
Control Package	6		
Pressure Sphere (1728 in^3)	18		
Nitrogen Gas	14		
Destruct System		10	
Payload Module		Variable	
Contingency		10	
NOSE FAIRING			84
*LAUNCH GROSS WEIGHT - Less P/L			34,343
AFT EXTENSION (Not part of 30' Vehicle; separated prior to first ignition)			974
Horizontal Surface Structure		368	
Vertical Surface Structure		368	
Operating Mechanism		96	
Aft Fairing		87	
Separation Mechanism (Sta 400)		55	
TOTAL WEIGHT DROPPED FROM CARRIER (Less P/L)			35,317

*CG at ignition; 136" from base

Moments of Inertia: 48725 slug ft^2 in pitch & yaw
2635 slug ft^2 in roll

WEIGHT SEQUENCE
Configuration B

LAUNCH GROSS (Less P/L Module)	34,343
Ignition Losses	9
FIRST IGNITION (Less P/L)	34,334
Propellant	20,730
FIRST BURNOUT (Less P/L)	13,604
First Stage Separation	3,357
Nose Fairing	84
Second Stage Ignition Losses	8
TVC Purge Fluid	3
SECOND IGNITION (Less P/L)	10,152
Propellant	8,880
TVC Fluid	92
SECOND BURNOUT (Less P/L)	1,180
Second Stage Separation	890
WEIGHT ON ORBIT (Less P/L Module)	290

CONFIGURATION C

Configuration C is essentially Configuration B plus a third propulsion stage for increased performance, resulting in a vehicle configuration approximately five feet longer than the 30 foot limitation placed on Configurations A and B.

The first stage is the same as Configuration B; the second stage is the same as Configuration B with the thrust termination parts deleted. The equipment bay and equipments are the same as in Configuration B. The third propulsion stage is mounted forward of the payload module and consists of a 36"-diameter spherical solid-propellant motor incorporating thrust termination provision and mounted backwards inside the nose aerodynamic fairing. After second stage separation, the entire third stage is re-oriented 180° prior to ignition.

CONFIGURATION C

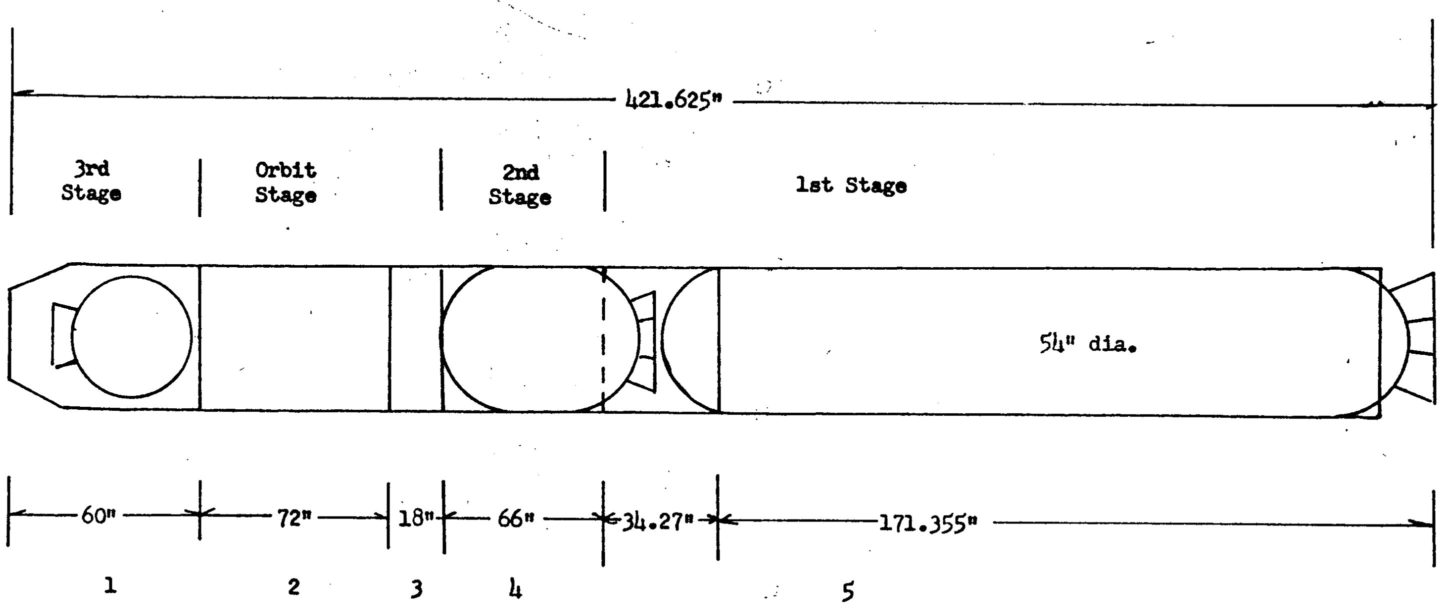

FIGURE 6

WEIGHT STATEMENT
Configuration C

3 propulsion stage version

FIRST STAGE			24,096
Propulsion		23,680	
Propellant	20,730		
Inert	2,950		
Ballistic Shell		117	
Electrical		110	
Flight Controls		154	
Contingency		35	
SECOND STAGE			9,868
Propulsion		9,500	
Propellant	8,880		
Inert	620		
Ballistic Shell		86	
Electrical Harness & Circuit		31	
Flight Controls		226	
TVC Hardware	106		
TVC Fluid	120		
Contingency		25	
THIRD STAGE			1,547
Propulsion		1,517	
Propellant	1,400		
Inert (12# expendable)	117		
Thrust Vector Control		30	
Hardware	15		
Fluid (12# expendable)	15		
ORBITAL VEHICLE (Less P/L Module)			350
Structure		110	
Equipment Bay	50		
Aft Separation Ring	10		
Motor Support (thru P/L section)	50		
Electrical		70	
Battery	26		
J Box, Switches, etc.	10		
Harness	34		
Guidance System		100	
Attitude Control System		50	
Nozzles, Valves, Plumbing	18		
Pressure Sphere (1728 in^3)	18		
N_2 Gas	14		
Destruct System		10	
Contingency		10	
NOSE FAIRING			84
*GROSS WEIGHT (Less P/L Module)			35,945

* CG at ignition; 151 in from base

Moments of Inertia; 76043 slug ft^2 in pitch & yaw
2815 slug ft^2 in roll

WEIGHT SEQUENCE
Configuration C

Propulsion Stage [illegible]

GROSS WEIGHT (Less P/L Module)	35,945
Ignition Losses	9
FIRST IGNITION (Less P/L)	35,936
Propellant	20,730
FIRST B/O (Less P/L)	15,206
First Stage Separation	3,357
Nose Fairing	84
Second Stage Ignition Losses	8
TVC Purge Fluid	3
SECOND IGNITION (Less P/L)	11,754
Propellant	8,880
TVC Fluid	94
SECOND B/O (Less P/L)	2,780
Second Stage Separation	883
THIRD IGNITION (Less P/L)	1,897
Propellant	1,400
TVC Fluid	12
Expendable Inert	12
THIRD B/O (Less P/L)	473
Motor Inert, TVC Hardware & Trapped Fluid	123
WEIGHT ON ORBIT (Less P/L Module)	350

PERFORMANCE

Payload capabilities and related performance parameters are summarized in the following table. Configuration B' is a version of B with improved motor characteristics. Propulsion system characteristics for configurations A, B and C are listed in Table II.

TABLE I

Performance Comparison

Polar Orbit - 80 Miles

Configuration		A	B	B'	C
Payload Wt. - lb.		635	340	610	1,010
Orbital Veh. Wt. - lb.		975	630	900	1,360
Launch Angle - deg.		27	38	38	33
Ignition Wt. - lb.		35,961	34,683	35,287	36,955
Propellant Wts - lb.	Stage I	26,000	20,730	21,830*	20,730
	Stage II	4,500	8,880	9,000*	8,880
	Stage III	--	--	--	1,400
Isp - sec.	Stage I	280	263.5	263.5	263.5
	Stage II	280	273.5	277.6*	273.5
	Stage III	--	--	--	305
Dimensions:	Dia/Depth	40	54"	54"	54"
	Length	30	30	30	35'

* The 270# increase for B' is due to:

Stage I, 1100 lb more propellant
626 lb less motor inert

Stage II, 120 lb more propellant
4.1 sec higher Isp

Launch Conditions

In all cases performance capability calculations are based upon first stage ignition at M = 3.0, 80,000 feet. Launch angle is the flight path angle (zero angle of attack) at first stage ignition which maximizes the orbital payload weight.

Free-fall time is assumed to be short so that the effects of the drop on flight performance are negligible. The aft extension with its stabilizing fins falls off at ignition so that it imposes no weight or structural penalty to the boost system.

Zero-lift or "gravity-turn" trajectories were computed on the IBM 7090 with a 3-degree-of-freedom, spherical-earth program. The final stage burn is at constant attitude relative to the local horizontal.

Configuration A

No satisfactory solid rocket motors of 40 inch diameter or less are available. (A brief examination of liquid-propellant systems indicated that a configuration sized to the length limitation of 30 feet was not feasible.) In conjunction with Lockheed Propulsion Company, booster systems were optimized to provide maximum stage impulse within the total length/depth limitations. The characteristics of these motors (two "bottles" attached side-by-side in each stage) are summarized in Table II.

This configuration will put a 635 pound payload on an 80 mile polar orbit (Figure 7). The launch angle is 28 degrees.

Configuration B

If the allowable diameter is increased to 54 inches, the Polaris A-3 motors will deliver 340 pounds into orbit (Figure 8). These conclusions are based on the performance of the -138 motors. However those of the -137 configuration can deliver 610 pounds into orbit; this configuration is referred to as B'. It is probable that much of this 270 pound gain can be realized. In the first stage, where the increment is approximately 190 pounds, the manufacturer is currently redesigning the motor to provide the desired increased propellant weight with a lower inert weight.

The launch angle for either B or B' is 39 degrees. If the launch angle is restrained to 30 degrees, the penalty in payload weight for turning the flight path through 9 degrees should be small.

Certain "trade factors" and penalties were computed for configuration B. Increased specific impulse will yield 8 and 15 lb/sec for stages I and II, respectively. A horizontal launch will reduce the payload weight by 200 pounds, and a M = 2.0 launch will cost approximately 200 pounds.

Configuration C

A three-stage vehicle (ML 470) has been proposed for a Navy satellite system. This vehicle uses the A-3 for the first two stages with a 36 inch spherical-motor third stage. This motor delivers 305 seconds specific impulse with beryllium additives. This propulsion stage on top of configuration B exceeds the 30 foot length limit by approximately 5 feet. Configuration C has a payload capability of 1010 pounds with a 33 degree launch angle (Figure 9).

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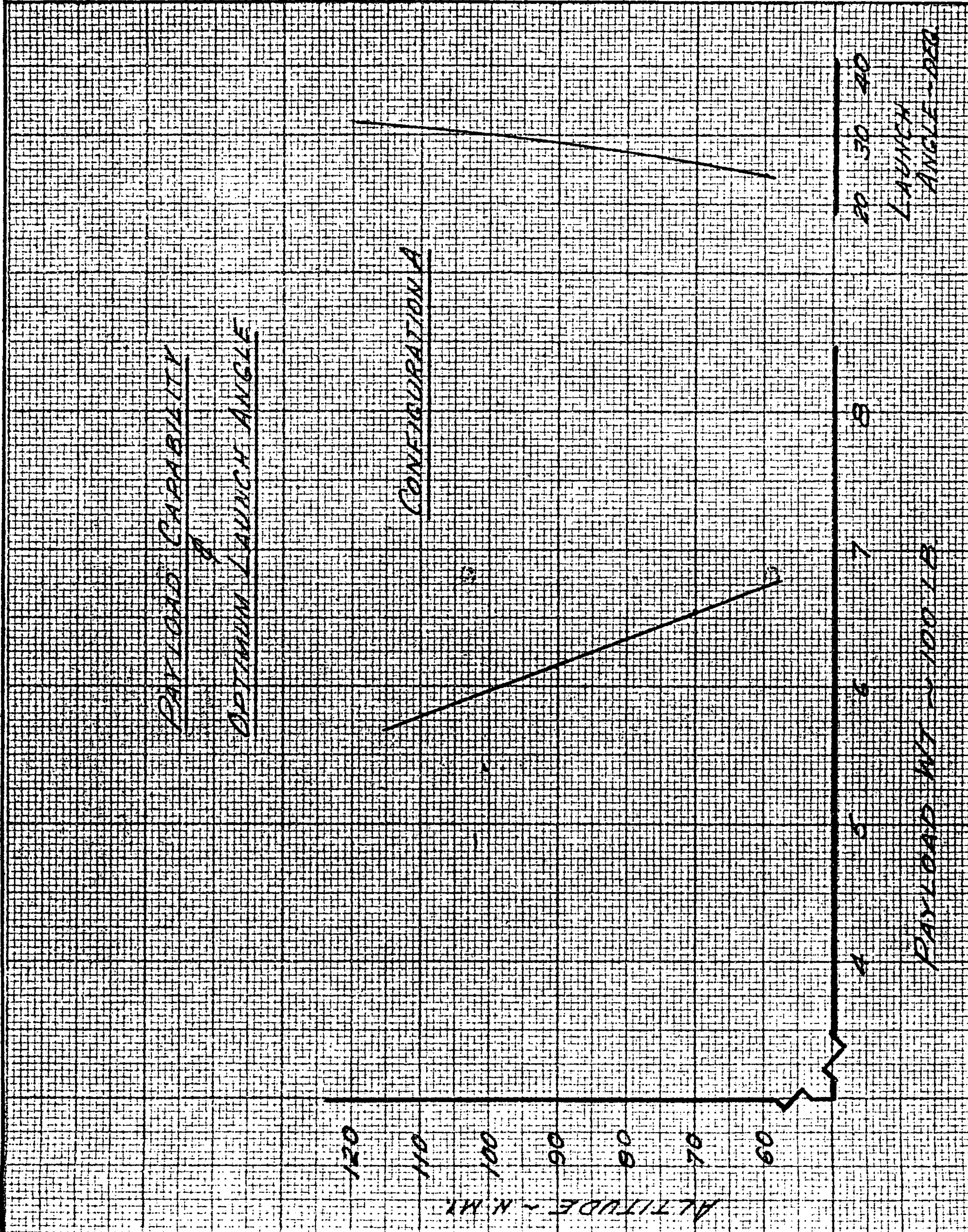

FIGURE 7

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FORM LMSC

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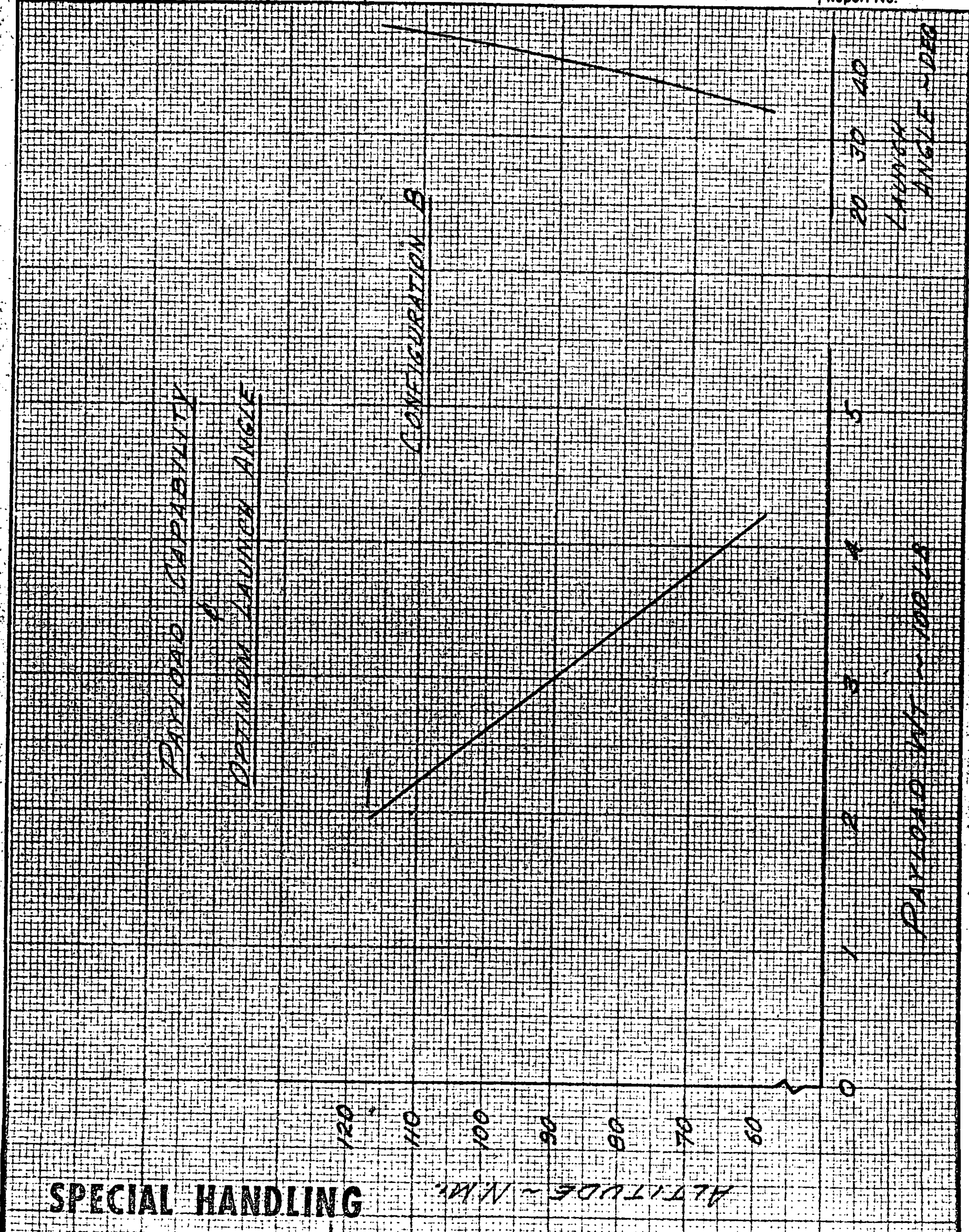

FIGURE 8

SPECIAL HANDLING

FORM LMSC 558A-2

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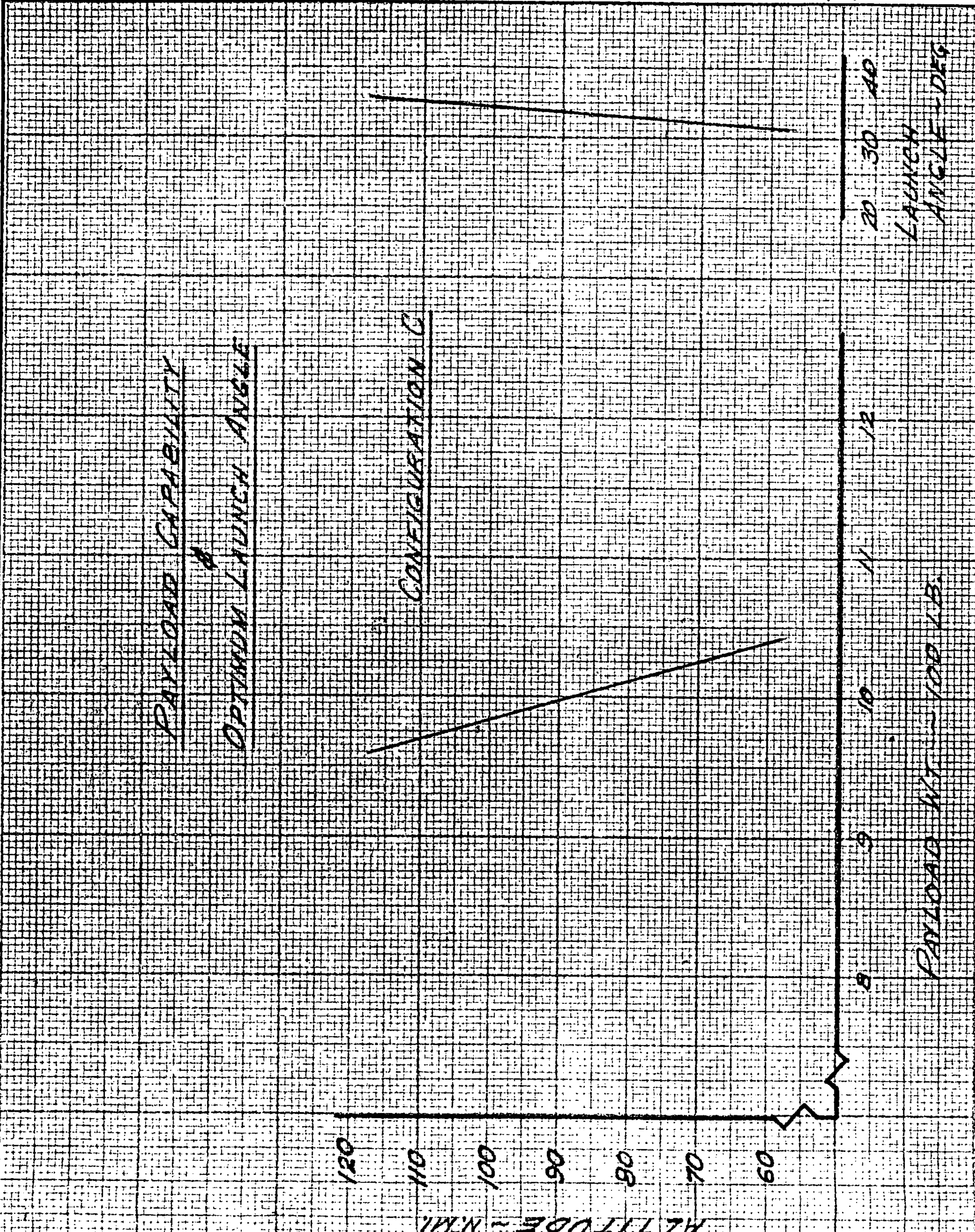

FIGURE 9

SPECIAL HANDLING

FORM LMSC 55

TABLE II

MOTOR CHARACTERISTICS

Config.	Stage	Motor Mfgr.	T Ave. Lb.	I_T X 10^{-6} Lb-Sec	I_{sp} Sec.	W_T Total Weight	W_P Propellant Weight	W_P/W_T Mass Fraction
A (1)	I	LPC Proposed	137,000	7.28	280	28,432	26,000	.815
	II	LPC Proposed	160,000	1.26	280	5,000	4,500	.90
B	I	Aerojet	80,000	5.46	263.5	23,680 (3)	20,730	.88
	II	Allegany Ballistics Lab.	31,000	2.43	273.5	9,580 (2)	8,880	.92
C	I	Aerojet	80,000	5.46	263.5	23,680 (3)	20,730	.88
	II	Allegany Ballistics Lab.	31,000	2.43	273.5	9,500	8,880	.93
	III	(ML-470 Proposed)	10,700	.43	305.	1,517 (2)	1,400	.92

NOTES:
1. Total (2 Motors) each Stage
2. Includes Thrust Termination
3. Rotating Nozzles
4. Vacuum

GUIDANCE & CONTROL

Guidance

The guidance system steers the vehicle into a precise, near-circular, low-altitude orbit. Thrust direction is controlled through attitude commands and termination is by engine cutoff command. On orbit, the system provides accurate attitude references for the control system and proper attitude and timing references for the capsule re-entry phase.

The initial launch conditions are M = 3.0 at 80,000 feet, flight path angles up to 40 degrees, and the proper direction to achieve the desired orbit inclination angle. Flight time from carrier take off to vehicle launch is assumed to be less than two hours. The carrier is assumed to have a high quality inertial navigator to provide the space vehicle guidance system with initial conditions.

Requirements for the guidance system are:

1) high orbit-injection accuracies
2) attitude references on orbit as follows:
   a. vertical vehicle yaw axis
   b. 90° yaw rotation immediately after injection
   c. maintain a yaw reference such that the vehicle pitch axis is lined with the earth relative velocity vector
   d. provide appropriate signals to the payload section
3) provide proper references for de-orbit as follows:
   a. capsule pitch and yaw attitude
   b. a timing initiation signal compatible with the objective of minimum impact dispersion

System Description

An inertial guidance system will be mechanized to have a minimum number of components by utilizing as much of the carrier and ground equipment for flight preparation as possible, and by combining the functions of missile guidance and control. (Figures 10 and 11).

Vehicle Inertial Guidance Equipment

A three gimbal inertial platform system will be used. Because of the 90° yaw maneuver after injection the platform is mounted with the outer gimbal aligned with the roll axis rather than the conventional alignment with the pitch axis. Ascent pitch maneuvers are about the middle gimbal.

The platform would have the following components:

1) 3 single degree of freedom (or 2-two degree of freedom) gyros.
2) 3 accelerometers
3) 3 gimbals, torque motors, and resolvers (or digital pickoffs)

The platform electronics house the signal and power amplifiers and gyro signal resolvers to derive proper platform gimbal torquing signals. In addition, the guidance computer attitude commands (see "Guidance Equations") will be brought into this package and compared with the gimbal angles to derive autopilot errors. The guidance computer will take initial condition inputs, guidance constants, and acceleration (or integral of acceleration) and compute the necessary attitude commands, thrust termination, and timing functions to meet the mission requirements. The OV computer will be a simplified special purpose computer with constants being derived either by the ground or aircraft computers.

Initial Conditions

Of prime importance to an accurate inertial guidance system are the initial conditions of velocity and position in platform coordinates. The system determining the initial conditions must know the vehicle reference coordinate system (alignment), or it assumes a reference coordinate system and the vehicle's platform must be aligned accordingly. Since the carrier is assumed to have accurate position and velocity information from its own navigator, the initial vehicle platform alignment is the only concern. Several techniques of alignment can be considered.

1) Accelerometer feedback - the accelerometer signals from the airplane inertial platform can be compared to the vehicle's accelerometers to derive signals for errecting the vehicle platform. Under steady state horizontal flight conditions this method can only align the vertical. If an accelerated or decelerated flight profile is used all three axes can be aligned by acceleration comparison.

2) Alternate techniques would be to align azimuth and vertical on the ground just prior to launch, optical alignment of azimuth via windows between carrier and vehicle platforms or a separate star tracker on the vehicle's platform to align azimuth. Detailed error analyses are required before the most suitable method can be determined.

Guidance Equations

The guidance system generates the steering command signals and terminates thrust in such a manner that all mission requirements are accomplished. In addition, various command signals are required during the orbital phase.

Ascent Phase

Either of two general methods of guidance currently under study are feasible for the mission. These are systems employing: 1) explicit guidance equations, and 2) delta guidance (or modified "Q" guidance) equations. For ascent the explicit guidance system is straightforward and requires a minimum of pre-launch calculation. However, the number of equations to be mechanized (approximately 24) results in a computer weight penalty estimated at 20 pounds as compared with the computationally simpler delta guidance schemes.

Orbit Phase

In orbit the guidance system generates the timing signals to initiate payload and recovery functions and provides attitude signals to the vehicle by means of the platform gimbals. A timing correction to include the effects of uncompensated cut-off impulse is provided.

Error Analysis

A preliminary error study was made of the inertial guidance system for airlaunch to near-circular orbits. The method of analysis was the statistical combination of error values due to the various sources assuming independence.

Error Sources

The error sources considered were: 1) the initial condition errors existing at the start of the boost phase, and 2) the guidance component errors and computer error within the space vehicle.

The initial condition errors in the inertial reference system in the launch vehicle were taken as follows:

Launch Vehicle Initial Guidance Errors (3σ)

Given (after 2 hours)	Assumed
position: 1 mile	velocity: 1 ft/sec
azimuth alignment: 100 sec.	pitch alignment: 12 sec.
	roll alignment: 12 sec.
	altitude: negligible

The guidance system in the space vehicle was assumed to be a current-day high-performance inertial package with the following characteristics:

Space Vehicle Guidance Errors (1σ)

accelerometer scale factor	2×10^{-5}
accelerometer bias	2×10^{-5} g
accelerometer construction misalignment	3 sec.
gyro random drift	.005 deg/hr
gyro mass unbalance	.01 deg/hr/g
gyro anisoelasticity	.003 deg/hr/g^2

An associated guidance computer was assumed with an error (3σ) equivalent to 1 ft/sec at orbit injection.

Error at Injection

The combination of the above errors when applied to an 80 mile orbit give the following 3σ orbital uncertainties:

eccentricity	$\pm$ .00012
period	$\pm$ 2.7 seconds
inclination	$\pm$ .050 degree
perigee-apogee difference	5160 feet

Excluded was the effect of uncompensated thrust impulse at cut-off. This is estimated as equivalent to a 1 ft/sec injection velocity error.

Error at Retro-fire

The projection of the injection errors through one complete orbit give the following altitude, velocity, and flight path uncertainties (3σ).

altitude	$\pm$ .26 miles
velocity	$\pm$ 4.5 feet/second
flight-path angle	$\pm$ .011 degree
in-track position	$\pm$ 11 miles
cross-track position	$\pm$ .06 miles (maximum of 3 miles at range angles of 90 and 270 degrees from launch.)

Attitude Reference Accuracy

The total attitude reference error combines the effects of the auto-pilot, syncho-resolver, stepper motor resolution, platform drift, initial alignment, equation approximations and conditions of non-standard flight. The total attitude error after one orbit can be reasonably estimated at $\pm$ 0.2 deg. excluding control system effects.

Equipment Characteristics

Weight and Power

For inertial guidance equipment, which can be made available within a year, the weight and power characteristics are estimated as follows:

Weight (total)	= 100 lbs. (conservatively)
Power (ascent)	= 360 watts
Power (orbit)	= 76 watts

Environmental Performance

The strict accuracy requirements of the mission demand that the performance characteristics of the guidance system be maintained in the in-flight environment. Selection of equipment will be directed toward those components and assemblies whose performance has been proven in flight or sled testing as well as conventional laboratory environmental testing.

ATTITUDE CONTROL

The attitude-controls system establishes thrust vector control of the booster first stages during the programmed ascent. At first stage burnout, that stage is separated by means of retro rockets and the second stage coasts to apogee of a transfer ellipse. During this coast phase the second stage is controlled about all three axes by means of the pneumatic (nitrogen) reaction jet attitude control at the front of the orbital vehicle (OV). During second stage burning, control about all three axes is achieved by means of fluid injection into the engine nozzles. Upon achieving orbital velocity, the second stage engines are shut down, and the OV, which contains the guidance platform and pneumatic attitude control system, is separated from the second stage booster. The OV is yawed through 90° to accommodate the payload, and the orbit phase is initiated. On orbit, the OV is controlled about all three axes by means of the pneumatic attitude control system. While on orbit, the OV is programmed through a yaw maneuver to compensate for the earth rotation. Upon completion of the orbit phase, the OV is oriented to the appropriate attitude for capsule separation, and the recovery phase is initiated. During the various phases of flight, the vehicle attitude commands and rate information are derived from the guidance system.

Boost Stages

First stage thrust-vector control is achieved by means of the hydraulically operated engine gimbal systems. The four engines are gimbaled so as to give three axis control. Second stage operation is similar except that fluid injection into the engines is used to achieve thrust vector control. The block diagrams for attitude control for the boost stages are shown in Figure 10, where the block labelled servo systems represents the hydraulic servo for the first stage or the flow control system for the second stage.

During the ascent phase, the computer-timer has stored within it the required trajectory information. This information is applied to the attitude control system as command signals by having the computer drive a motor-resolver system. The output of the motor-resolver is compared with the gimbal angles of the platform and the resultant signal is used as the attitude signal. In order to damp the vehicle motion, angular rate information is required. This rate information may be obtained from the platform servo torquer current or by phase leading the attitude information. Use of the torquer currents m ay provide a cleaner signal; however, LMSC has had considerable success with phase lead techniques so that both approaches will be considered. The attitude and rate information is summed in the flight controls electronics, and the resultant error signal is amplified and applied to the hydraulic servo (or flow control valve) to direct the engine thrust of the four engines to obtain control of all three axes.

Orbital Vehicle

The attitude control system for the OV is used during the coast to second stage ignition and is used continuously on orbit.

After injection into orbit, the OV is separated from the second stage and is yawed through 90^{o}. The platform gimbals then are orthogonal to the body axes. This orientation simplifies determination of attitude excursions while on orbit. Thereafter, a constant roll rate is imparted to the OV such that it is aligned to the local horizontal in pitch and roll throughout the orbit; a programmed yaw command is imparted to OV as a function of latitude. The block diagram for attitude control of the OV is shown in Figure 11. This system utilizes the platform for attitude information and the pneumatic reaction system of the Agena D.

Because of the orientation of the OV and body axes, pitch and yaw attitude information is directly obtainable from the platform gimbals. The computer-timer together with the motor-resolvers generates a programmed signal for roll so that the OV is given a continuous roll rate to align the OV to the local horizontal throughout the orbit flight. In a similar manner, a preprogrammed yaw command is imparted to the vehicle to compensate for the earth's rotational velocity.

Vehicle angular rate information for damping is obtained from the platform torquer signals and applied to the flight controls electronics.

At the end of orbit operation, the vehicle is pitched down and stabilized prior to initiation of the capsule recovery sequence.

Flight Controls Electronics

The flight controls electronics sums the signals of the motor-resolver and platform (both attitude and angular rate) and applies the resultant error signals to the appropriate pneumatic reaction jets for vehicle stabilization. A deadband circuit of 0.1 deg. (about all three axes) is used in the flight controls electronics to conserve attitude control gas during orbit operation.

The amplified error signal is applied to the pulse electronics circuits (in the flight control electronics) which generates the pulse commands to the pneumatic reaction jet valves.

Pneumatic Reaction Control System

The pneumatic reaction control system consists of pulse thrustor valves, two stage pressure regulator, and gas storage tank.

The pulse thrustor valves are simple solenoid valves with a rapid response (less than 0.025 sec. pulse time), and essentially no leakage.

The two stage pressure regulator is assumed for operation of the thrustors at two force levels: 10 lb. for use during ascent, and 0.5 lb. on orbit.

ATTITUDE CONTROL BLOCK DIAGRAM FOR BOOST STAGES

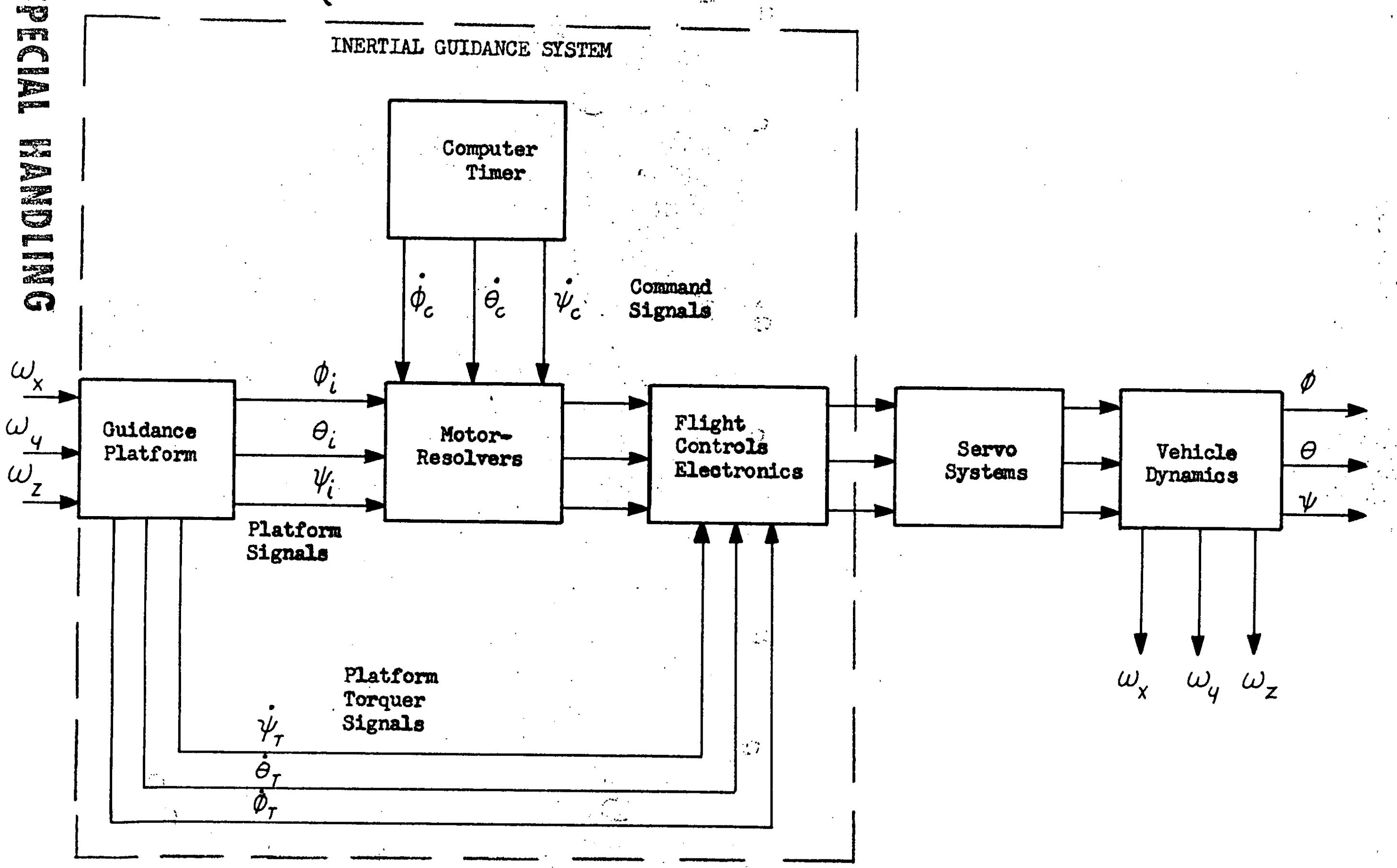

Figure 10

ATTITUDE CONTROL BLOCK DIAGRAM FOR O.V.

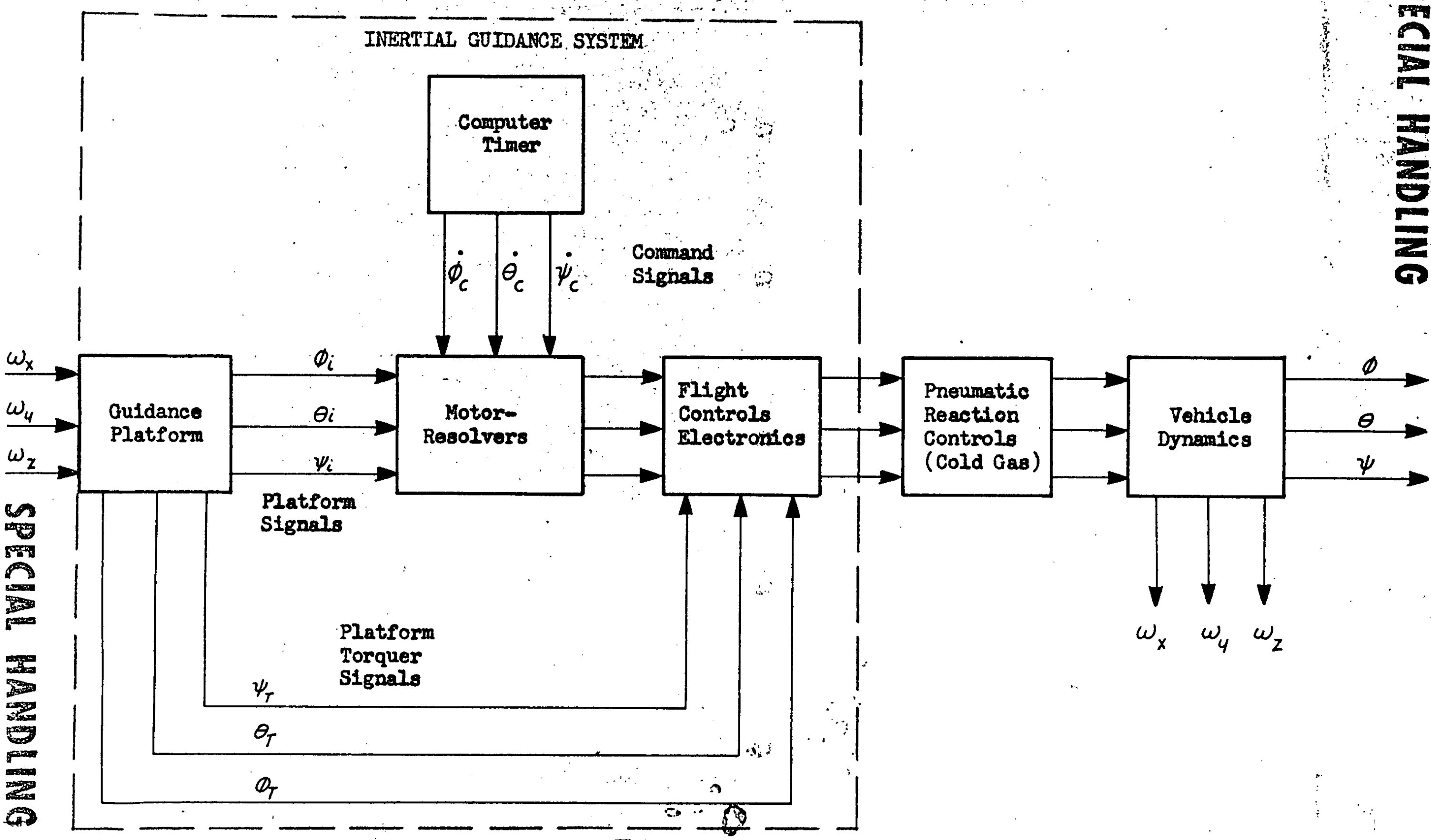

Figure 11

SPECIAL HANDLING

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STRUCTURES & LOADS

Design Criteria for Configuration B

The vehicle is designed for the normal flight loading conditions determined by the trajectory and vehicle configuration. In addition, vehicle structural integrity is assured for the added conditions imposed by the proposed method of launch. The new, or added, design conditions and the estimated design load factors are:

Condition	Design Load Factor - g's Longitudinal	Lateral	Vertical
Ground Handling	± 2	± 1	± 2
Mating With Carrier	± 1	± 1	± 2
Take-off With Carrier	± 2	± 1	+ 2
Flight With Carrier	± 2	± 1	+ 2
Release From Carrier	± 1	± 1	+ 2
Tail Jettison	± 1	± 1	± 2
Landing With Carrier	(Not Applicable)		

Compliance with the ground handling and mating load condition restrictions is achieved by appropriate design of ground handling equipment and by control of procedures.

The "Take-off" and "Flight With Carrier" conditions will be dependent upon the carrier condiguration and performance; however, the loadings assumed above are considered appropriate.

The "Release From Carrier" condition assumed the vehicle dropped away from the carrier at a trimmed condition with an angle of attack below eight degrees.

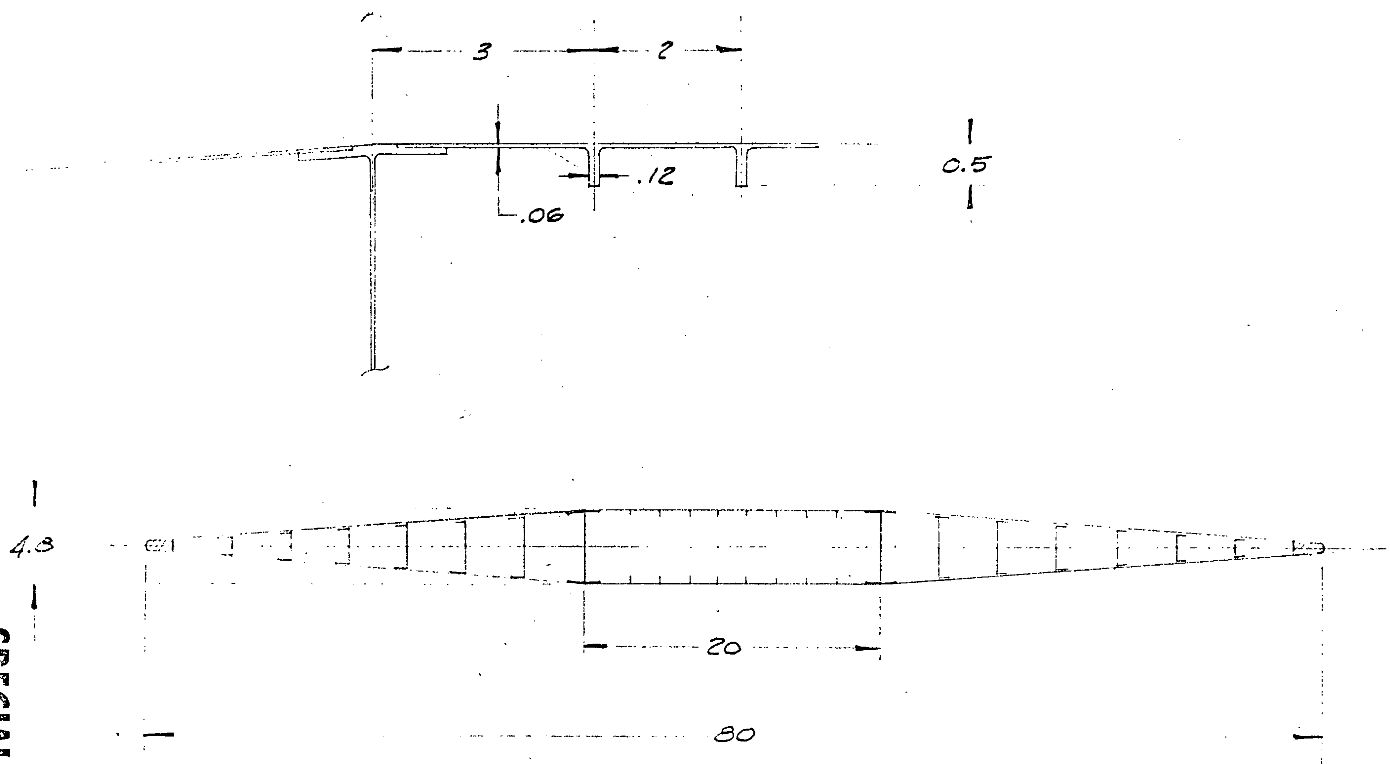
The tail jettison mechanism will be designed such that the above listed loads on the vehicle will not be exceeded.

It is assumed that the vehicle will be jettisoned from the carrier prior to landing in the event of an aborted vehicle launch; consequently the vehicle design need not incorporate landing condition loads.

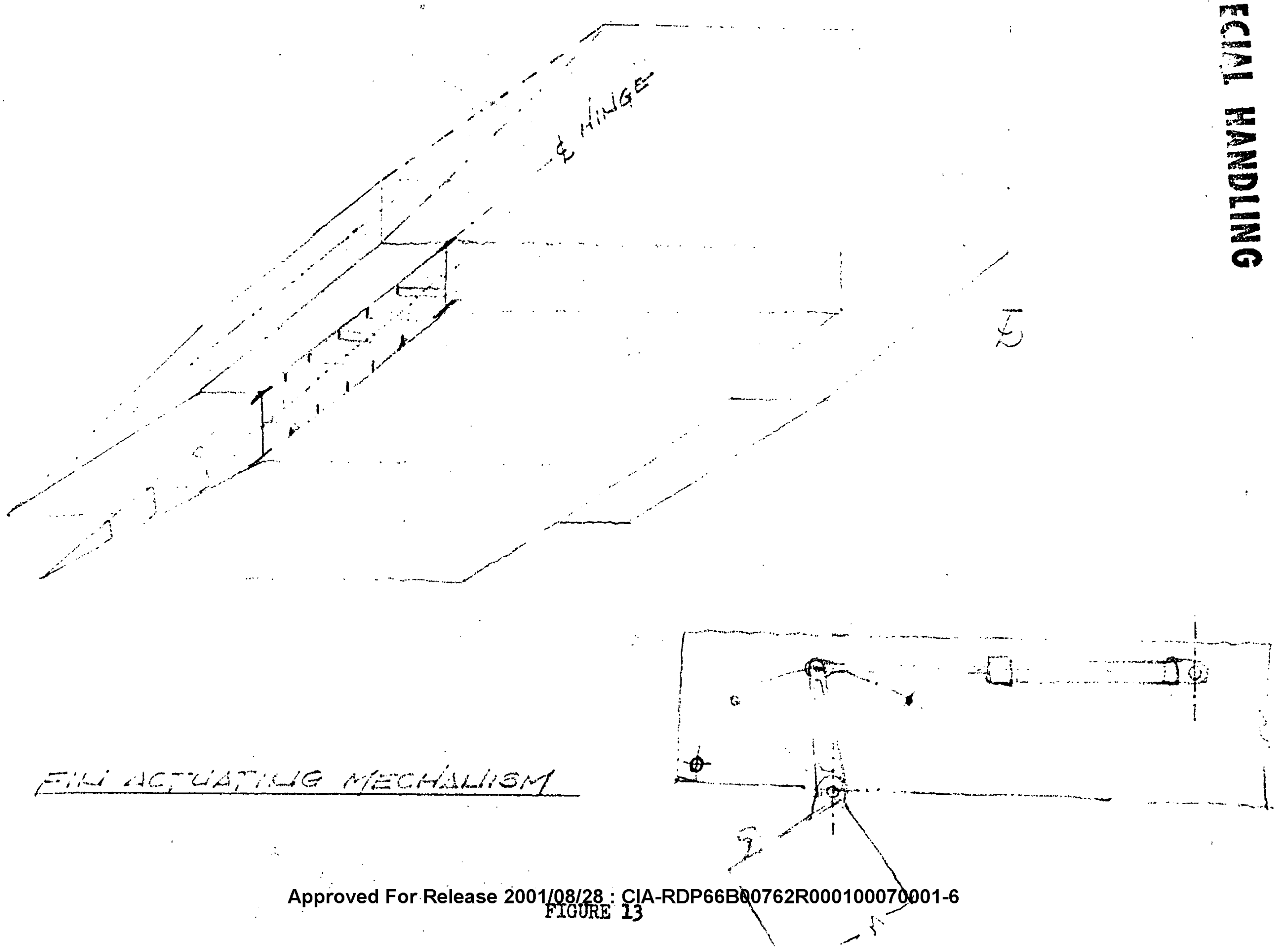
The exact configuration of the proposed aft extension structure (including fairing, horizontal tail surface, and vertical tail surfaces) will be contingent upon wind tunnel tests and interference characteristics of the carrier. In the proposed preliminary tail configuration, the surfaces are rectangular in planform and of 80 inch chord. The required span of the horizontal tail surface will be from 10 to 15 feet. The dual vertical tail surfaces are hinged to the lower surface of the horizontal tail at or near the outboard ends in order to permit folding of the vertical fins for required ground clearance during take-off.

A double wedge airfoil section with a 20 inch wide structural box and shear webs of machined titanium or steel alloy is proposed. Leading and trailing edge skins are of titanium or steel with light gage 'C' section spanwise webs as shown in Figures 12 and 13.

The forward main support fitting for attachment of the vehicle to the carrier is located in the first stage/second stage interstage structure and consists of a 54" diameter ring and a hookup fitting as shown in Figures 14 and 15. The aft support fitting is incorporated into the aft extension structure.

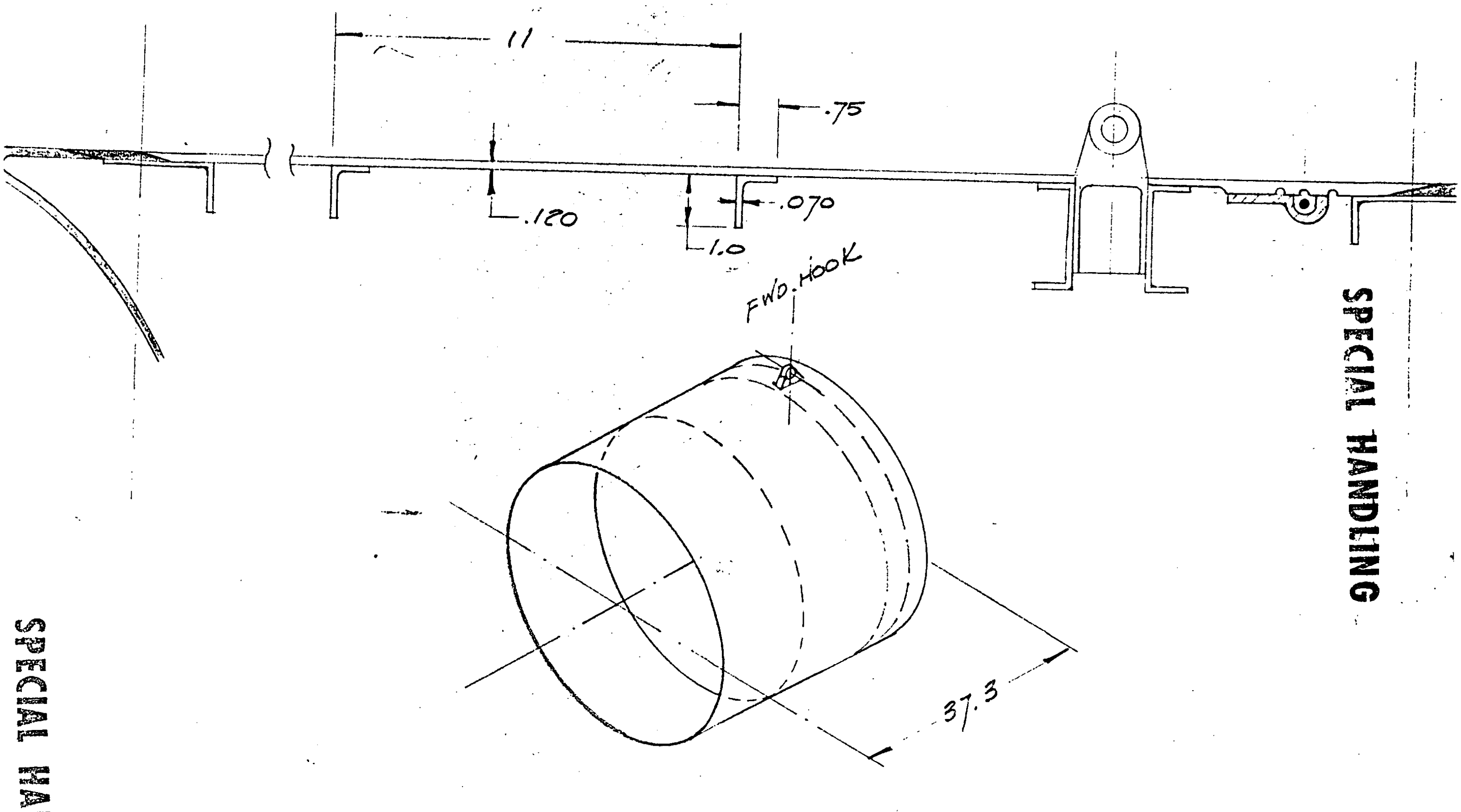


FIN & STABILIZER SECT. (RC-130 Ti)



FILM ACTUATING MECHANISM

FIGURE 13



INTERSTAGE STRUCTURE
MAG. ALLOY

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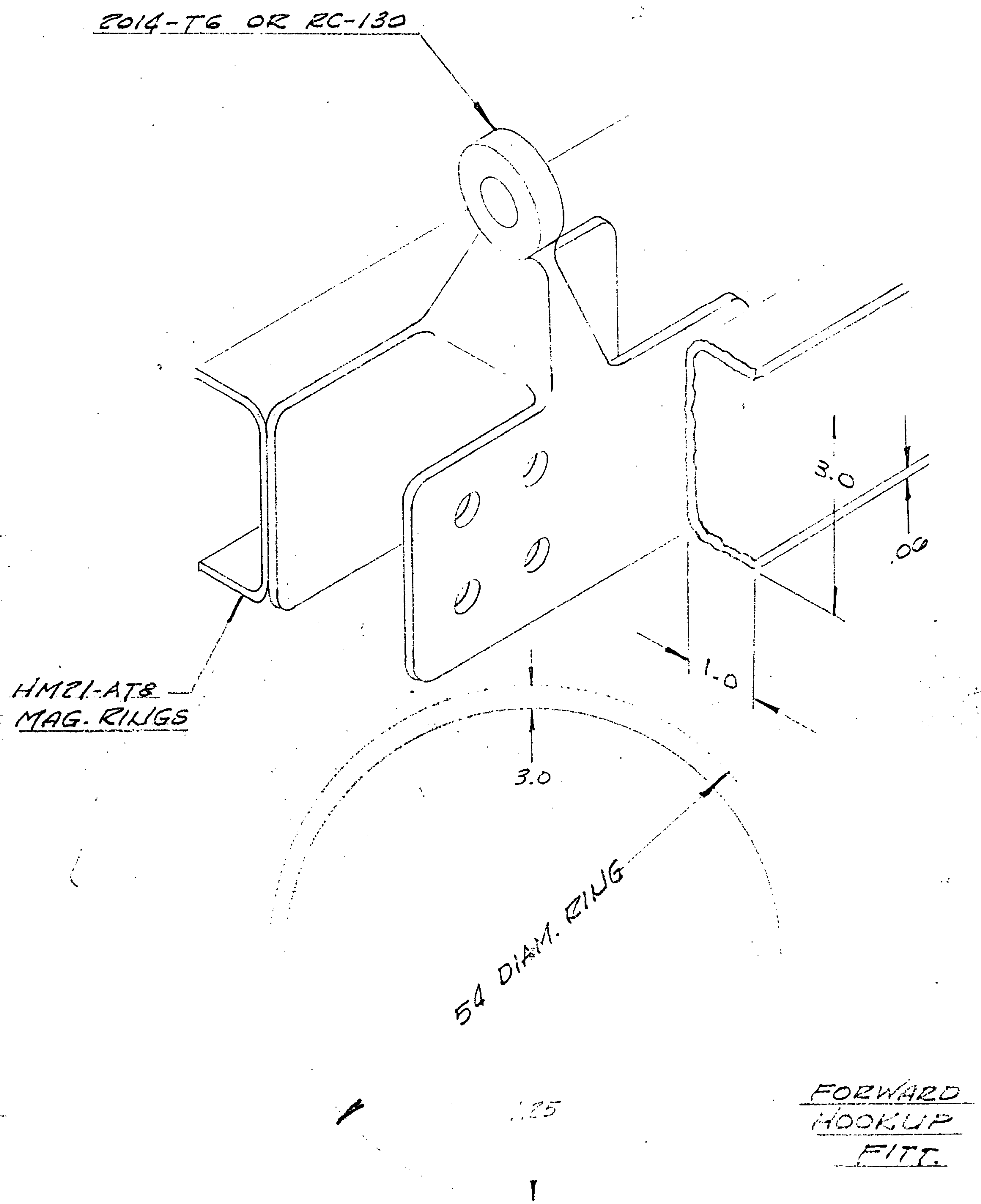

FIGURE 15

SPECIAL HANDLING

ADDENDUM TO CONFIGURATION C

This study primarily was directed toward Configuration B since it meets the 30-foot length limitation. However, the half-ton payload capability of Configuration C has evoked considerable interest which had led to further study on this vehicle.

Effect of Launch Speed and Angle

The ascent-trajectory technique was altered from the optimum-launch-angle, gravity-turn method to one employing a pitch up to 10° angle-of-attack during first-stage burn. The results of computer runs for a range of Mach numbers between 2.2 and 3.2 are shown in Figure 16. It is interesting to note that the payload capability for an 80 mile orbit dropped less than 10 pounds at $M = 3.0$ with a drop in required launch angle from 33° to less than 18°.

Impact Dispersions - Effect of Errors at Launch

For the given space vehicle guidance system, the effect of the launcher guidance uncertainty on re-entry capsule impact dispersion was studied. The pitch and roll alignment errors were increased to 30 seconds of arc while velocity errors were increased from 1 to 5 and 10 feet per second. The errors (uncertainties) at retro fire and corresponding impact dispersions are compared in Table III. The parenthetical numbers in the last column are the results of the original study discussed in the Guidance and Control, and Payload Sections.

Temperature Environment Prior to Launch

The A-3 propulsion stages are structurally critical at temperatures above 200°F and payload systems are designed for initial launch temperatures near 70°F. The ideal cruise Mach number, therefore, is near 1.5. If aircraft requirements dictate higher speeds, the orbital vehicle must be cooled; if the cruise Mach number exceeds approximately 2.0, the first and second booster stages also will require cooling.

Techniques for protecting the orbital vehicle are briefly covered in discussion of the spaceframe in the Payload System section. The external blanket or shroud imposes no payload penalty on the space vehicle. If cooling is required, liquid cooling within the orbital vehicle walls is more desirable than internal air conditioning.

The critical regions on stages 1 and 2 are the bonds between the propellant liner and motor case, and the bonding of the interstage attachment rings to the fiberglass motor case. The attachment bonds could be protected by coolant coils in the interstage regions with little weight penalty. The motor cases will require insulated or cooled external blankets which are separated from the space vehicle just prior to launch. The third-stage can be protected by insulation inside the nose cap.

Parachute Stabilization During Free-Fall

A supersonic parachute has been suggested as an alternative to the stabilizing fins on the aft extension. A 20-foot diameter chute will provide an adequate static margin of 12 feet without imposing excessive loads on the structure. Vehicle oscillations are less than 1 cps with a half-amplitude of approximately 5 degrees. Load factor on the chute and structure is 1.4 g's; the total axial deceleration due to chute and body drag and gravity is 2.0 g's.

The effect of "time-on-the-chute" is shown in Figure 16A. Initial speed has been increased to M = 3.2 to partially compensate for the effects of the chute so that the basic (zero time) capability has increased 60 pounds to 1070. Separation distance is along the flight path and assumes constant airplane speed after launch. Free-fall times should be kept to a minimum, consistent with vehicle safety, since it penalizes capability at the rate of nearly 20 pounds per second, and increasing free-fall time imposes more adverse heat-load problems on the chute design.

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EFFECT OF LAUNCH CONDITIONS

CONFIGURATION C

10° ANGLE OF ATTACK DURING FIRST STAGE BURN
0° (GRAVITY TURN) DURING SECOND STAGE BURN

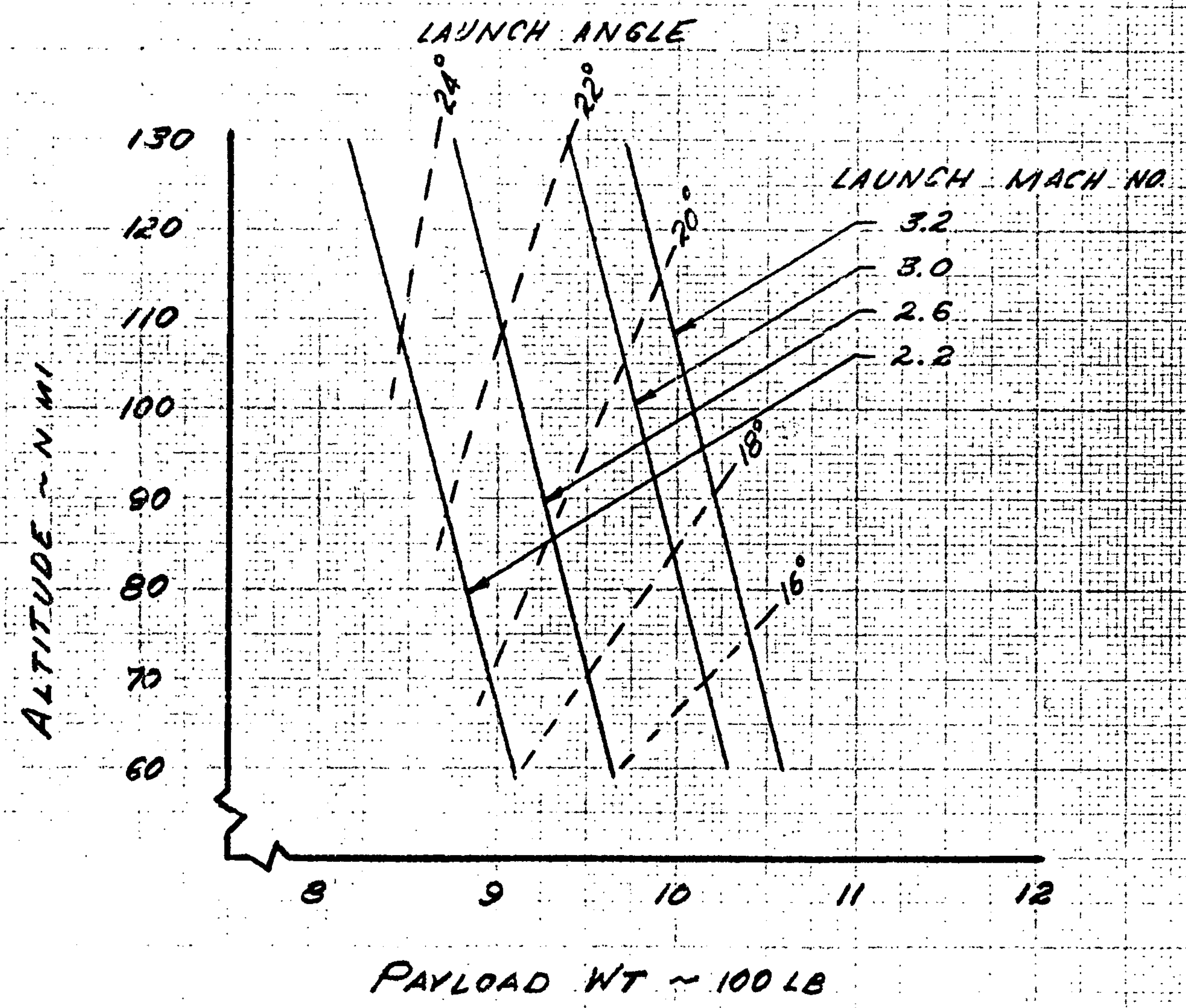

FIGURE 16

TABLE III

EFFECT OF LAUNCH CONDITIONS ON IMPACT DISPERSIONS

Launch Vehicle Initial Conditions:	Velocity	ft/sec	1	5	10	(1)
	Pitch/Roll Alignment	seconds	30	30	30	(12)
Errors (3σ) Retro-Fire:	Altitude	mi	.30	.30	.30	(.26)
	Velocity	ft/sec	4.5	6.5	11.9	(4.5)
	Flight-path angle	deg	.012	.012	.012	(.011)
	In-track position	mi	11	16	29	(11)
Impact Dispersions (3σ):	Due to Guidance	mi	15	20	33	(15)
	Total Dispersion - Uprange	mi	21	24	36	(20)
	- Downrange	mi	25	28	38	(25)

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EFFECT OF TIME-ON-CHUTE

CONFIGURATION C
20 FT DIA CHUTE
M = 3.2
1070 LB PAYLOAD CAPABILITY (t=0)

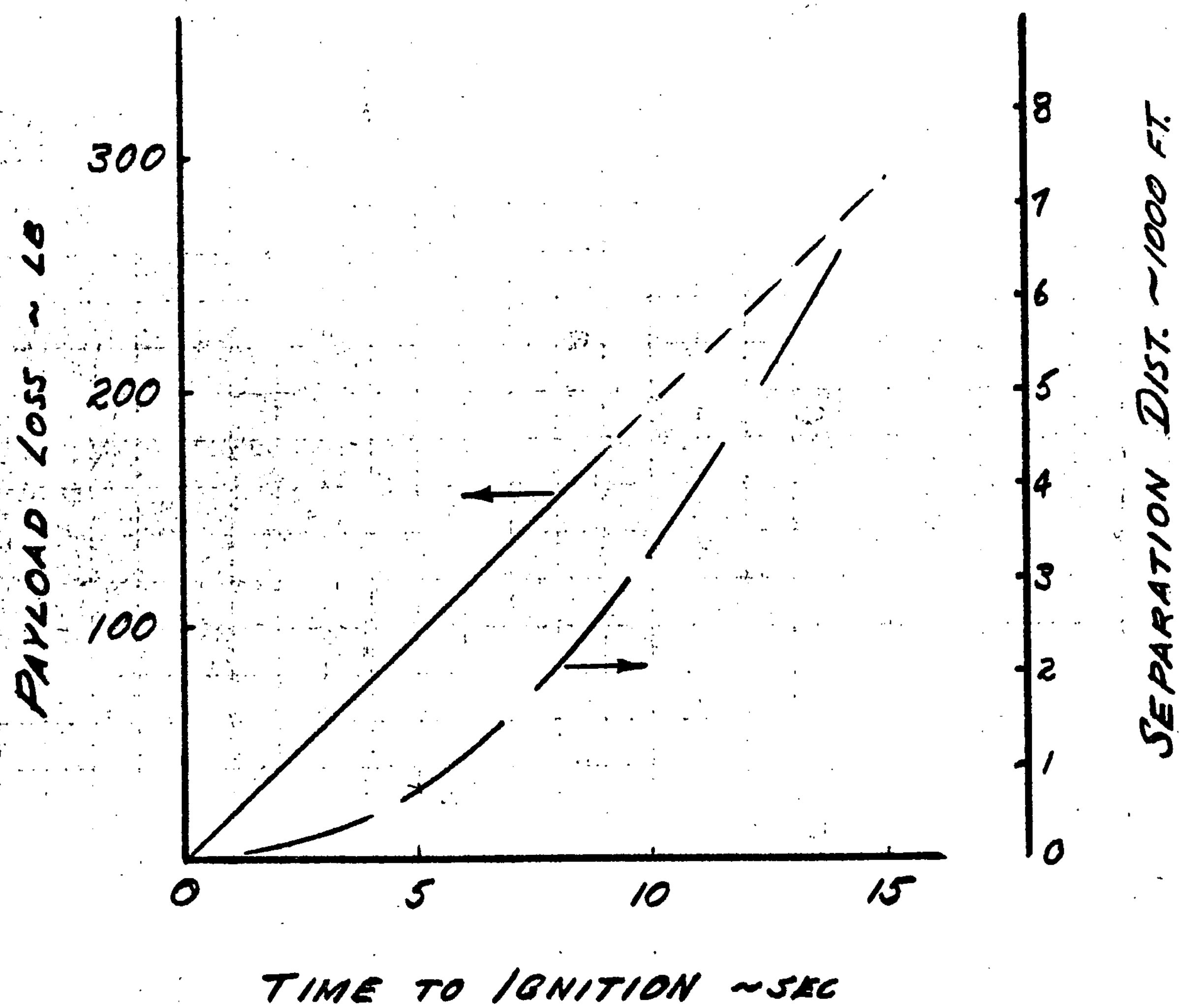


FIGURE 16A

PAYLOAD SYSTEM

GENERAL

The orbital vehicle will consist of an equipment module to which is attached a payload section. The payload section will consist of a panoramic camera and associated cassette, recovery system, payload power supply, and payload space structure.

Three types of panoramic cameras have been considered, each of which is a logical evolution of the existing and proven Corona system. In fact, Configurations I and II are direct modifications of this subsystem, and Configuration III is a direct application of a proposed single lens stereo camera subsystem. Primary effort has been directed toward the utilization of the camera subsystem described as Configuration I which is compatible with vehicle Configuration B/B'.

The recovery subsystem, common for all three payload configurations, is being developed for and will be qualified under the ML-470 Program (S^2).

The space structure subsystem will provide for camera subsystem installation, recovery subsystem installation, payload power supply, system intercabling, and auxiliary equipment required to provide proper support, physical integration and orientation of the camera and recovery subsystems. The space structure subsystem will be compatible with the vehicle configuration with which it is associated.

The payload system weights for each configuration are:

Configuration I = 456 lbs.

Configuration II = 300 lbs.

Configuration III = 700 lbs. (est.)

CONFIGURATION I

The hardware assembly with this payload system consists of:

a. Camera Subsystem - The camera subsystem shall consist of one Corona M_2 type camera with associated cassette. The characteristics of the camera are:

Type -	Fixed film panoramic
Lens -	f/3.5 Petzval with 40" focal length
Scan Angle -	60° (40° if vehicle design dictates)
Shutter -	focal plane, interchangeable slits for exposures of 1/250 sec., 1/500 sec., 1/1000 sec.
IMC -	Lens system moves in opposite direction relative to motion of vehicle
IMC Rate -	proportional to V/H setting
Film -	SO-132

The camera will be designed to mount a supply spool containing 700 feet of 5" wide 3.2 mil film. Based upon a 41.2 inch (60°) scan format length and 4.5 inch format width, this amount of film is capable of photographing approximately 155,000 square miles of the earth's surface at the operationally specified location. If the scan angle is reduced to 40 degrees, the 40 inch camera will be capable of covering 146,000 square miles.

The ground resolution of the total system operating at 80 nautical miles altitude is expected to be:

a. Medium Contrast (6.3:1) 3.5 feet

b. Low Contrast (2:1) 4.0 feet

b. Re-entry Capsule Subsystem - The re-entry capsule subsystem, developed for the ML-470 Program, consists of an ablative/structural shell, (a short cylindrical section coupled to a flared skirt) retro-rocket, spin assembly, power supplies, parachutes, beacon and flashing light assembly, wiring harnesses and associated hardware.

c. Structure Subsystem - The structure will consist of an elongated fairing section which will provide for camera installation, re-entry capsule installation, power supply, system inter-cabling, and the auxiliary equipment required to provide proper support, physical integration and orientation of the camera and re-entry capsule subsystems.

WEIGHT - CONFIGURATION I

RECOVERY VEHICLE		76
INSTRUMENT		240
AIR FRAME		91
Barrel	58	
Nose Cone	33	
	91	
A/P ELEC		39
Harnesses	23	
Sig. Cond.	2	
Commutator	3	
Programmer	8	
Misc.	3	
	39	
PAYLOAD		10 (5" x 7.00" Dia.)
TOTAL		456 LB.

SPECIAL HANDLING

CONFIGURATION II

This configuration will provide a light weight, early operational capability by utilizing one Corona Mural camera which has a 24" focal length f/3.5 Petzval lens and which, under the same conditions as Configuration I, will offer the following performance:

a. Medium Contrast (6.3:1) = 5.0 feet

b. Low Contrast (2:1) = 6.0 feet

This camera has a scan angle of 70 degrees and uses 70 mm film. At an orbit altitude of 80 nautical miles, and using 700 feet of film, this camera is capable of photographing 180,000 square miles. Swath width photographed is 92 nautical miles. This configuration will be similar to the ML-470 payload shown in Figure 18.

SPECIAL HANDLING

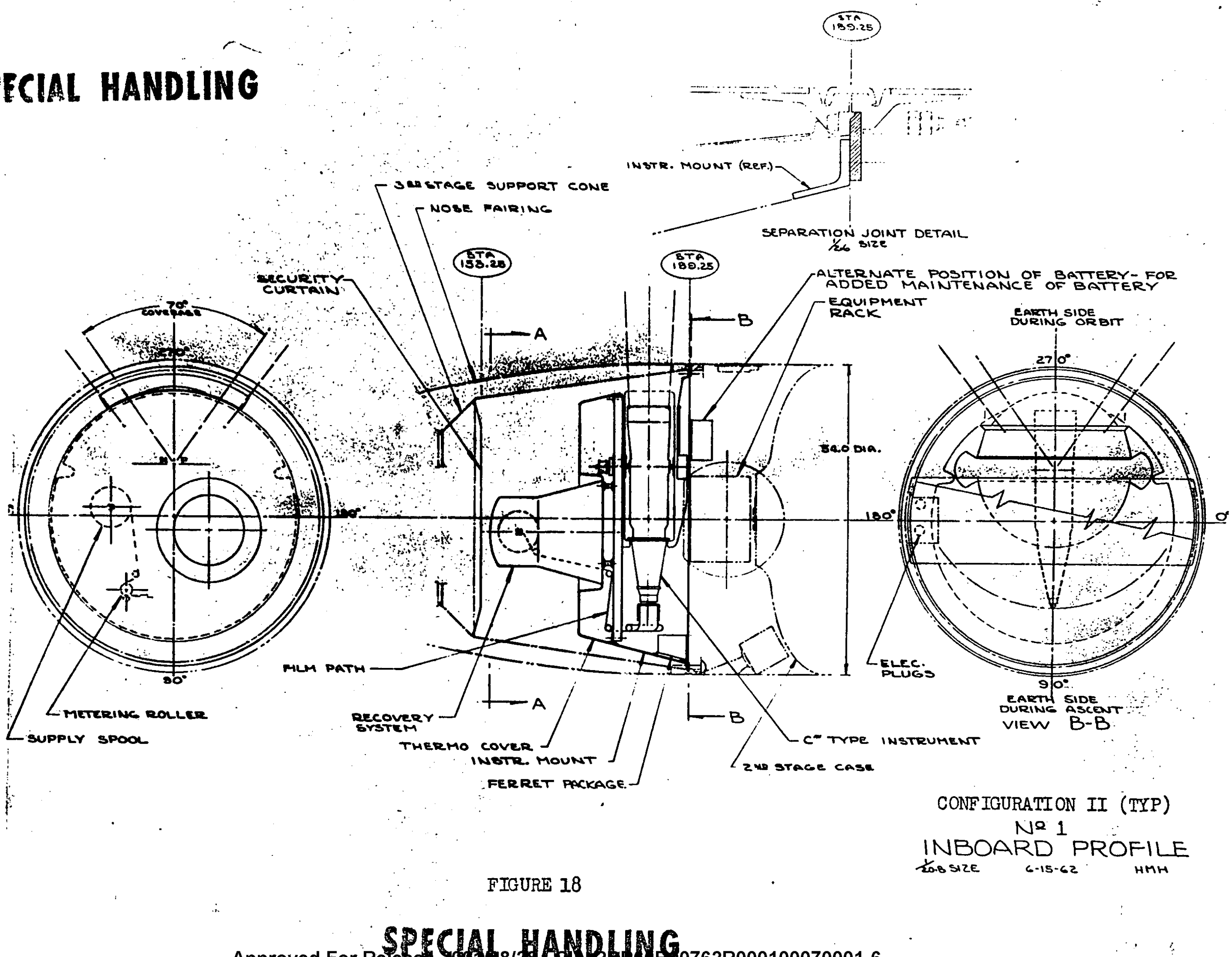


FIGURE 18

SPECIAL HANDLING

CONFIGURATION III

This configuration is a single lens stereoscopic panoramic camera built around the 40 inch focal length Petzval lens used in Configuration I. This single lens stereo camera design is being investigated by ITEK Corporation for WADO.

Basically, the camera consists of two platens mounted at the proper distance and angular orientation to obtain 30 degree convergent oblique stereoscopic configuration. The lens is then positioned by moving mechanisms so that it alternately scans the forward looking and aft looking platens.

Independent film supply spools, film metering drives, and cassettes are provided. Image motion compensation is obtained by an IMC cam mounted on the rotating lens. The lens barrel actually describes an elongated ellipse as it scans each platen.

For this configuration, two 700 foot spools of film would be provided. Esitmated weight of this configuration is approximately 700 pounds. From a performance viewpoint, this camera should have a ground resolution equal to the Configuration I camera, and in addition offers the increased advantage of presenting the third dimension which is a great aid in identifying and recognizing photographed objects.

SPACEFRAME

The effect of orbital temperature gradients are not considered to be significant due to the short orbital life (one pass, $1\frac{1}{2}$ hrs.) relative to long time constant of the payload. Pre-launch and ascent heating constitutes the primary thermal problem with cruise temperatures possibly being the greatest contributor. The adverse effects on the payload of high cruise Mach number can be offset by incorporation of any of the following techniques:

1. Liquid cooling within the vehicle structural walls; the coolant will be provided by the carrier aircraft.
2. Transpiration cooling techniques developed on previous programs.
3. Removable thermal blanket or shroud.

No instrumentation will be flown on operational vehicles. It is expected however that early flights will be instrumented to monitor environmental data instrument performance.

For reasons of security there will be no telemetry, so that the use of a tape recorder for recording instrumentation data on early flights would be required. The recorder may be either recovered or read out during pre-recovery acquisition.

RECOVERY SYSTEM

The recovery capsule, developed for the ML-470 Program, is employed directly. It is a light weight cylinder flare configuration (Figure 19) weighing 76 pounds. The capsule is separated from the orbital vehicle after it has been re-oriented to the proper retro attitude.

A retro velocity of approximately 1300 ft/sec is imparted, after capsule spin up, by a low dispersion (less than 1%) motor. The motor case remains with the capsule to prevent the possibility of debris landing in the recovery area. Atmospheric deceleration of the spinning capsule is conventional, with the heat loads being absorbed by ablation. In order to minimize dispersion due to wind drift, parachute release occurs when the vehicle has descended to the lowest practical altitude.

The re-entry capsule is designed for air-snatch retrival with added capability of withstanding surface impact.

A SARAH beacon and a flashing light assembly will be employed as recovery aids.

RECOVERY SUBSYSTEM
PRELIMINARY WEIGHT STATEMENT

	LBS	LBS	TOTAL
1. Search & Acquisition Aids:			
a. Beacon (SARAH) & antenna		.75	
b. Beacon coder (signature)		.31	
c. DC to DC converter		.35	
d. Flashing light		.1	
e. Flashing light controller		.25	
		1.76	1.76
2. Sequencing & Deployment equipment:			
a. Recovery programmer & timer		.5	
b. Pyro ejectors (2) for 'chute cover		.18	
		.68	.68
3. Parachute System:	LBS		
a. Air Recovery version	8.3		
4. Structure:			
a. Recovery system & retro support		2.0	
b. Power supply containers		2.0	
c. Cassette encapsulator		.75	
d. Shell		2.0	
e. 'Chute cover (ejectable door)		.50	
f. Ablative material (estimate)		19.3	
		26.55	26.55
5. Payload & Payload Components:			
a. Payload		5.0	
b. Cassette motor		.2	
c. Cassette		.50	
d. Film cutter and water seal		1.0	
		6.7	6.7
6. Re-entry:			
a. Retro rocket		24.0	
b. "g" switches (2)		1.0	
		25.0	25.0
7. Power: Approx 15 A/H (12 hrs)			
a. Batteries		5.0	
b. Harnesses		2.0	
		7.0	7.0
Sub-total			67.69
Plus 3a (Air recovery)			8.3
Total			75.99

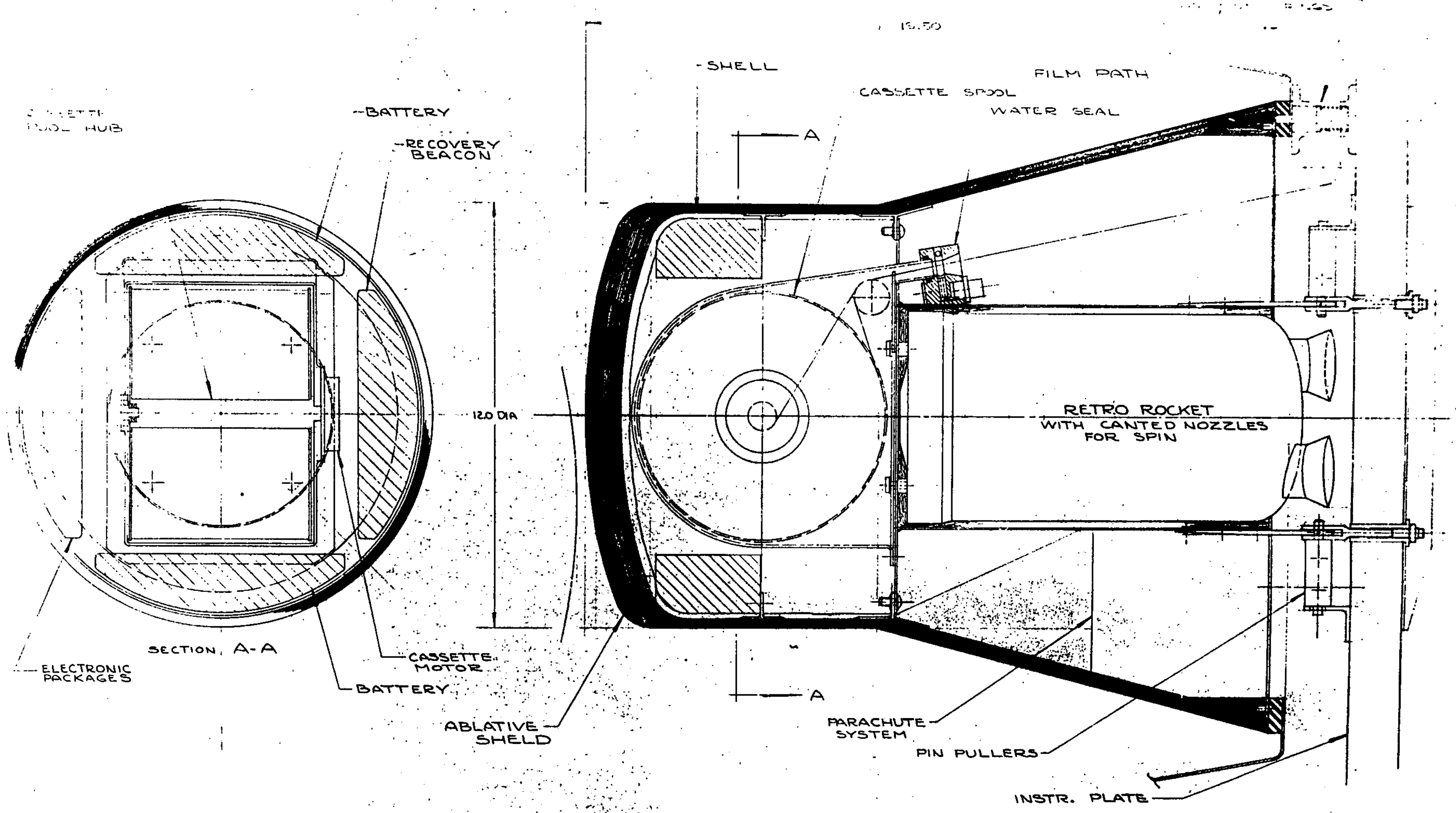


FIGURE 19

RECOVERY SYSTEM

Approved For Release 2001/08/28 : CIA-RDP66B00762R000100070001-6

RE-ENTRY IMPACT DISPERSIONS

The predicted re-entry dispersion is 20 miles uprange and 25 miles downrange, 3σ. Cross range dispersions are small, of the order of 5 to 7 miles. "Impact" is defined as the altitude of parachute opening. The relative contributions are as follows:

Effect	Dispersion-miles uprange	Dispersion-miles downrange
1. Orbital uncertainties after 1 pass	15	15
2. Retro velocity including 1% impulse variation	13	19
3. Atmospheric drag uncertainty	5	5
RSS - 3σ dispersion	20	25

The dispersions will grow rapidly with increasing orbit passes due to the effects of orbital period uncertainty, guidance drifts, and orbital drag.

SPECIAL HANDLING

PAYLOAD PROGRAMMING AND MISSION ANALYSIS

INTRODUCTION

The capability of the guidance and control subsystem to establish accurately the orbital injection parameters, plus the shortness and relative simplicity of the flight mission makes the task of programming easier than that of present missions or more sophisticated proposed systems; there are fewer commands to be computed and issued.

Computer programming will have the flexibility of rapidly providing operational constants which will permit the selection of any target within the denied area. All programming requirements are within the state-of-the-art.

ANALYSIS OF THE ORBIT

Swath width is a function of altitude and instrument scan angle. In past studies sometimes a "target" has been spoken of as an area 50 mi. X 50 mi.; in present missions a "target" is a range of latitudes specified for a particular rev number.

Since the present capabilities of the proposed instrument and the orbital vehicle indicate a good balance between orbit lifetime, scale factor, and ground resolution at an altitude of about 80 n. mi., the swath widths for various scan angles are calculated based on this altitude.

Scan Angle	Swath Width
20°	28 n. mi.
30°	43
40°	58
50°	75
60°	92
70°	112 n. mi.

It is desirable, from several standpoints, to attain orbits which are as nearly circular as possible, so that the velocity and altitude over the target remain nearly constant. The v/h ratio for the above orbit is .159 ft/sec/yd. with a cycle rate of 1.8 sec for 60° scan angle. Present instruments, with a 70° scan angle and a 24" focal length having a cycle period of 2.4 sec when cycling to match a v/h ratio of .117, are capable of achieving these new parameters.

Figure 1 shows typical ground tracks for three different inclinations of prograde orbits, 85°, 60°, and 30°. Orbits shown were all designed to be recovered at Johnston Island.

Target selection is provided by orbit inclination. Period variation provides a very limited target selection capability especially at low orbit inclinations. Additionally, this technique imposes greater demands on mission programming procedures and equipments because of the varying eccentricities which would be required. Since the guidance system's maximum cross-track error is only 3 miles, maximum simplicity is achieved by variation of orbit inclination.

Once the target coordinates are selected, a computer program will be employed to give rapidly the solar elevation at that location as a function of various launch times for the mission date, for analytical use in exposure studies, selection of film type, and determining correct aperture or slit width for optimum exposure over the target area.

SPECIAL HANDLING

FLIGHT MISSION/PROGRAMMING

Computer programs will be developed to generate the mission ephemeris in order to provide the necessary operational parameters.

Because of short mission life, no real time commands will be used. All commands necessary for a complete mission will be computed in advance of the mission, recorded from the computer into the satellite borne program device and stored for execution at the proper time.

The only payload commands needed are:

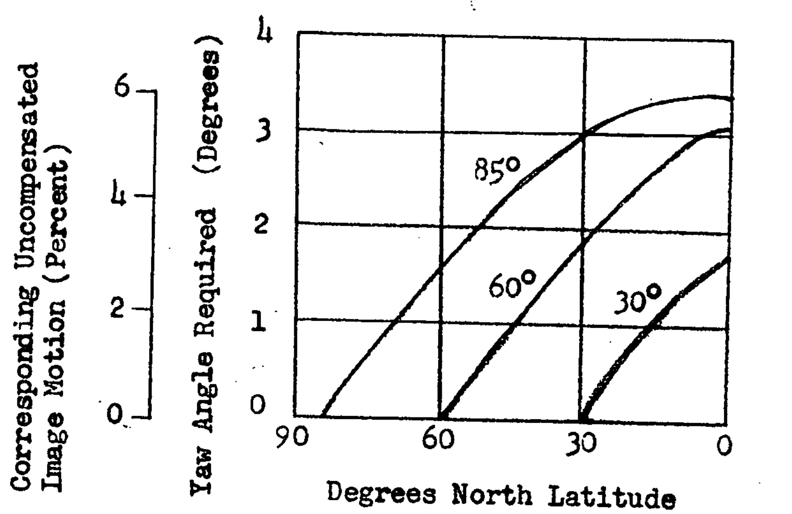
1. instrument on;
2. payload slew and instrument off;
3. recovery sequence initiate (provided by guidance computor)

The payload slew command is felt to be desirable to run any payload remaining after an operational pass into the cassette just prior to recovery. This serves the dual purposes of insuring a predictable re-entry vehicle weight as well as precluding an aborted mission in the event of water seal failure. A fixed cycle period will be computed and pre-set at any time up to launch aircraft take-off to provide a fixed V/H signal to the camera.

In order to fully realize performance capabilities of the proposed camera subsystem, vehicle yaw correction is required.

SPECIAL HANDLING

EFFECT OF INCLINATION ANGLE ON IMC



This will correct for uncompensated image motion due largely to earth rotation. Full alignment is provided by yawing the vehicle with respect to time. The IMC error from this source will be limited to less than 2%.

SPECIAL HANDLING

DISCUSSION

The configurations (B/B' and C) which utilize the Polaris propulsion system are discussed, since Configuration A requires an expensive motor development program and the 40 inch depth limitation no longer applies. The comparative table relates the payload weights with the 80-mile payload carrying capability of all configurations.

Vehicle Configuration \ Payload Configuration	I (456 lb)	II (300 lb)	III (700 lb)
A (635 lb)	x	x	
B (340 lb)		x	
B' (610 lb)	x	x	
C (1010 lb)	x	x	x

The 3-stage Configuration C is the only vehicle capable of carrying all three payloads, and therefore is the most desirable if the launch vehicle is capable of accommodating its 35 foot length. The three motors have been or will be developed and qualified for other programs, and an encouraging potential is provided for advanced payloads or retrograde orbits.

The primary vehicle of this study - the two-stage Configuration B/B' - is capable of carrying Configuration II for a certainty, and probably Configuration I, when the current motor improvement efforts are realized. It meets the 30 foot length limit and would be launched conventionally with a stabilized free-fall glide.

The system is launched so close to the Hawaiian Islands that it should be possible to compromise airplane cruise conditions in favor of the space vehicle temperature requirements. A cruise Mach number near 1.5 provides the ideal environment for the vehicle and payload systems. Higher cruise speeds will impose critical thermal conditions upon structural components and will require that active cooling techniques be employed.

The vehicle is completely silent; payload and vehicle guidance programming is performed on the ground prior to take-off. The guidance system will turn on the payload at entrance into the area of interest and will control the time of re-entry for minimum impact dispersion. Maximum cross-track error is 3 miles which is insignificant for the payload under consideration, and is satisfactory for any narrow swath spotting system which might be contemplated.

Re-entry impact occurs inside a narrow strip 45 miles long, and often in daylight. These conditions suggest the possibility of recovery by helicopter, possibly operating from shipboard.

PROS AND CONS

Advantages and disadvantages of the system described herein are discussed with respect to nine factors which are believed to be the most significant. (The order of discussion is not related to significance.) The figures in parentheses are an attempt to give a numerical rating. Numbers used are 1, 2, and 3; 1 being least desirable. On this basis, the system rating is 22 points out of a possible 27.

SPECIAL HANDLING

Vulnerability (3) - Vulnerability up to launch is low. Fighter protection can easily be supplied from take-off to launch.

Secrecy (1) - It is inconceivable that take-offs would not be observed. Knowledge of the launch point could probably be protected. However, this is of little significance since the enemy would soon become aware of the recovery point from observations within his own territory even if all other information were denied him. Thus the system is rated low in secrecy unless adequate take-off cover could be developed.

Response Time (2) - This is defined as the time from the establishment of a mission requirement to launch of the satellite. The time required for preparation to take-off and travel to the launch-point is sufficiently long to cause this factor to be rated medium.

Length of Cycle (2) - This is defined as the time from the establishment of a mission requirement to the start of processing of recovered material. The system is rated medium in this characteristic primarily because of its response time.

Target Selection Capability (3) - Aircraft mobility provides great flexibility in launch point and launch azimuth. Thus this factor is rated high.

Frequency of Operation (3) - Frequency of operation is limited only by the number of aircraft which can be provided and the rate of missile production. This factor is rated high.

Launch Point Flexibility (3) - When the target and recovery point have been specified, only a single degree of launch-point freedom remains. However aircraft can make full use of this freedom. Thus this factor is rated high.

Reliability (3) - All other factors being equal, reliability is highest when the shortest possible time exists between payload assembly and payload operation. Thus this system rated high in reliability.

Costs (2) - Analysis indicates that this system may be classified as a medium cost system.

A family of orbit traces which intersect at Johnston Island are mapped in Figure 1. The significance of the foregoing discussion is immediately recognized. It is seen that inclinations from 30° to 85° prograde will encompass the denied area with the exception of a small portion of Southwestern China and part of Eastern Europe. These latter can be covered by extending the range of prograde inclination angles a few degrees. It is also seen that a conjugate point exists in the southern hemisphere in South Africa. Orbit injection from sites other than these conjugate points require extension of the launch system capability; the degree depending upon distance from a conjugate point since the space vehicle must be injected into orbit along the appropriate azimuth lines if recovery is to be at Johnston Island.

Launch Location

The relationship of launch and recovery areas with the Hawaiian Islands is shown in Figure 2. Hickam A.F.B. is located less than 700 miles from the launch conjugate point. Though this distance is well within the capability of the launch aircraft, time of flight can be minimized by flying to the closest point of passage of the required azimuth line. These points are indicated on the figure and the corresponding distance from Hickam is shown in Figure 3. Low-azimuth launches would be restricted to the east of Hawaii. The cruise range to the launch site is between 250 and 600 miles.